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VOYAGER
SPACECRAFT Phase B, Task D

FINAL REPORT
OCTOBER 1967

Volume 4. Spacecraft Electrical Subsystems Definition

Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

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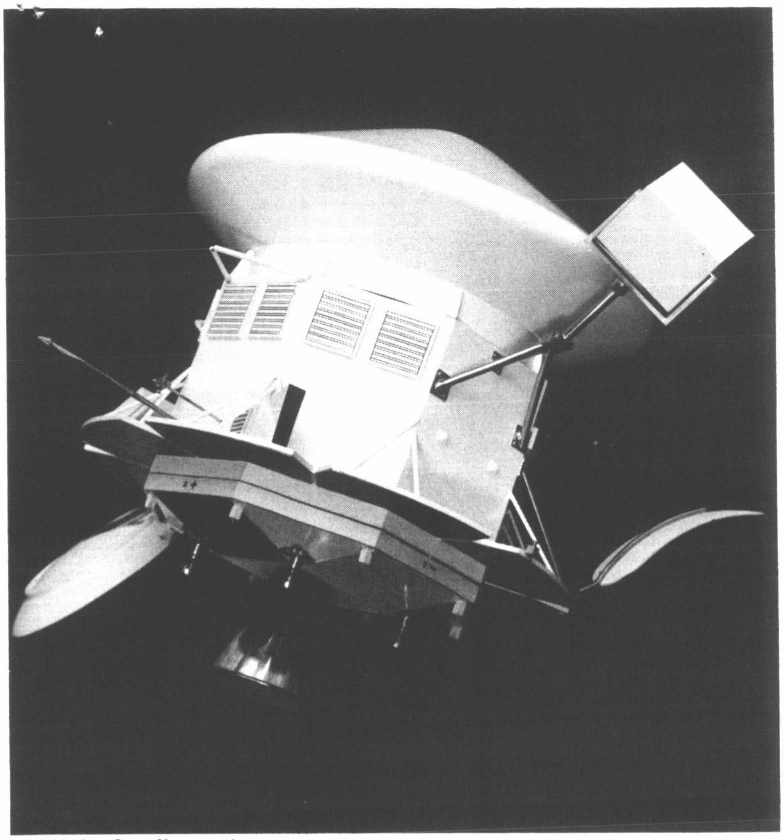
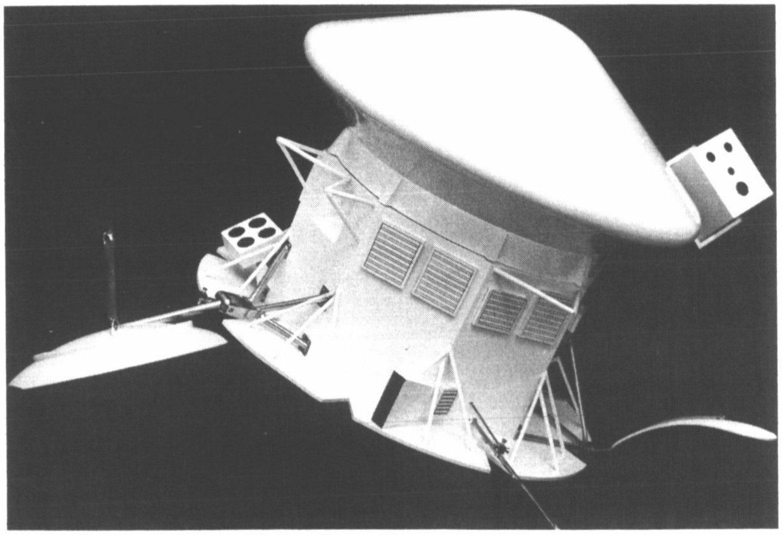
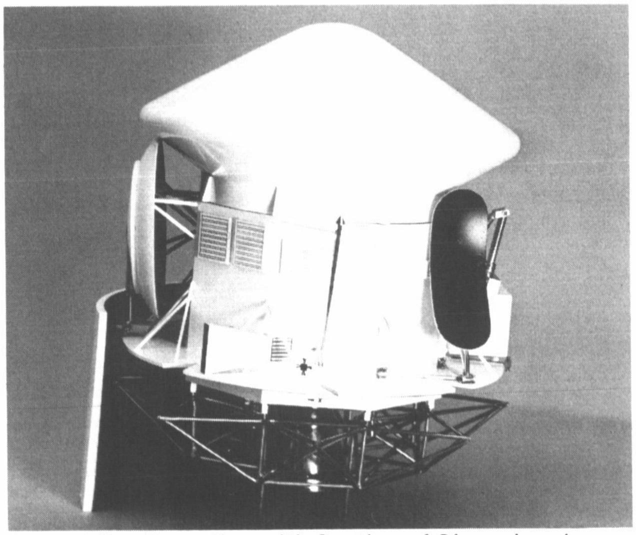


In-Flight Configuration

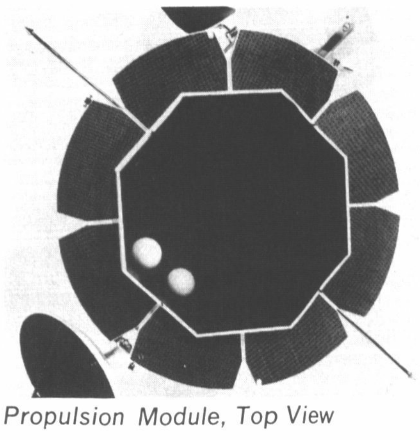
MODEL OF **TRW** RECOMMENDED **VOYAGER SPACECRAFT**



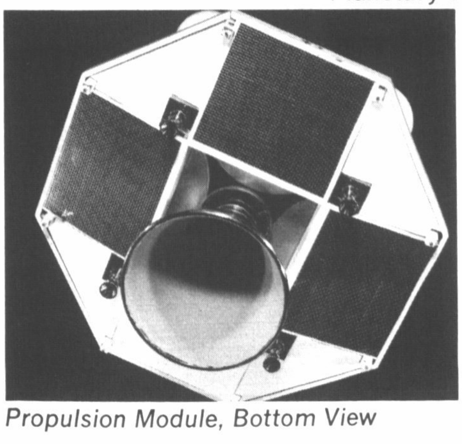
Opposite View In-Flight Configuration



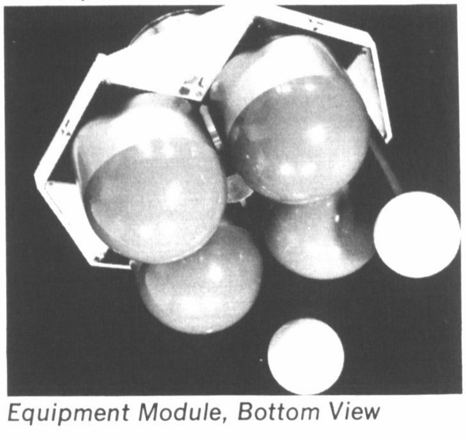
Stowed Configuration with Section of Shroud and Planetary Vehicle Adapter



Propulsion Module, Top View



Propulsion Module, Bottom View



Equipment Module, Bottom View

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VOYAGER

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CONTENTS

CONTENTS (Continued)

INTRODUCTION AND SUMMARY

This volume describes the electrical subsystems of the recommended spacecraft. All of the electrical subsystems are mounted on the equipment module (see Figure 1). Several closely related subjects are described in other volumes: the electrical GSE is described in Volume 7; the science payload and planetary scan platform are discussed in Volume 5; the various photo-imaging approaches appear in Volume 11; and the growth potential of the electrical subsystems is discussed in Volume 6.

The electrical subsystems are divided into three groups as shown in Figure 2:

a) Communication and Data Processing Group

- The Telemetry and Data Storage Subsystem receives science and measuring data; it multiplexes, formats and stores the data; it passes it on to the S-band radio subsystem.

- The S-band Radio Subsystem comprises the electronics for receiving and transmitting, including all antennae and their drives; it interfaces with the telemetry and command unit; it handles the tracking.

- The Command Subsystem detects and decodes the earth commands; it routes commands to the sequencer or selectively to the subsystems directly.

b) Guidance and Control Group

- The Guidance and Control Subsystem maintains spacecraft attitude at all times throughout the mission, and it controls the LMDE thrust vector. Optical and inertial sensors, reaction control system, and LMDE actuation are part of the guidance and control subsystem.

- The Computer and Sequencer stores the programs for all modes of operation; it receives new commands through the command unit; it steers the high-gain and the medium-gain antenna; it supplies orientation signals to the science subsystem; it gives cutoff to the LMDE and initiates the C-1 engine backup mode; it interfaces with all subsystems.

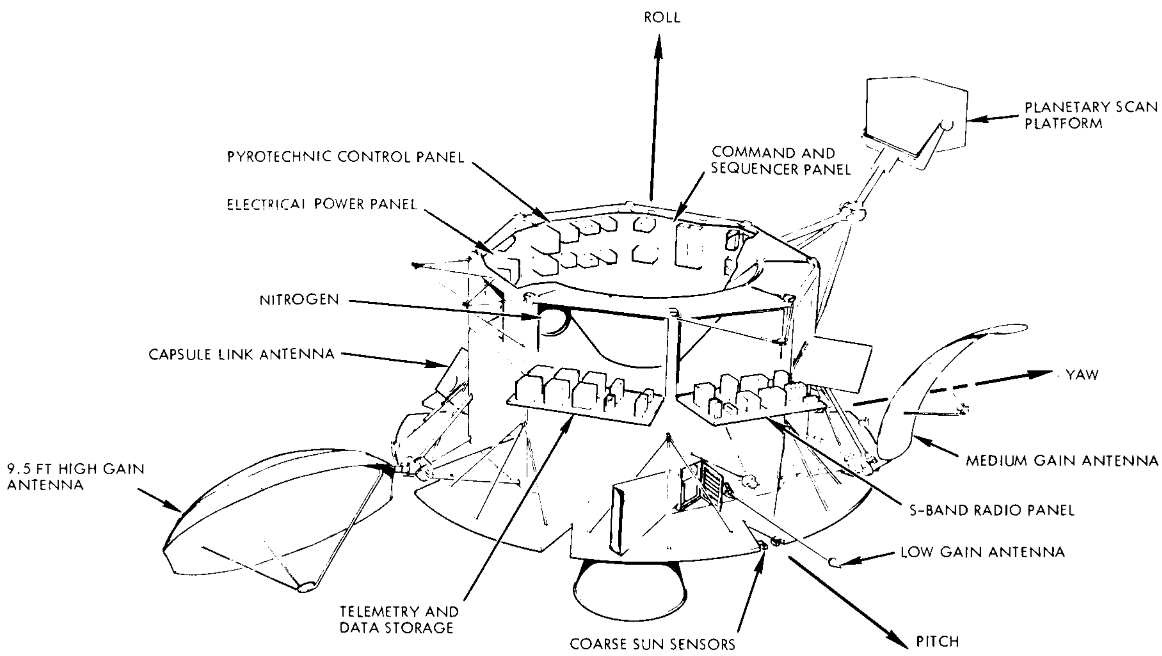


Figure 1

THE ELECTRICAL SUBSYSTEMS OF THE RECOMMENDED SPACECRAFT are mounted in the equipment module. Efficient access to the electronic modules of the subsystems is gained by use of hinged equipment panels. Additional space is available on these panels for growth in electrical subsystem performance for upgraded missions.

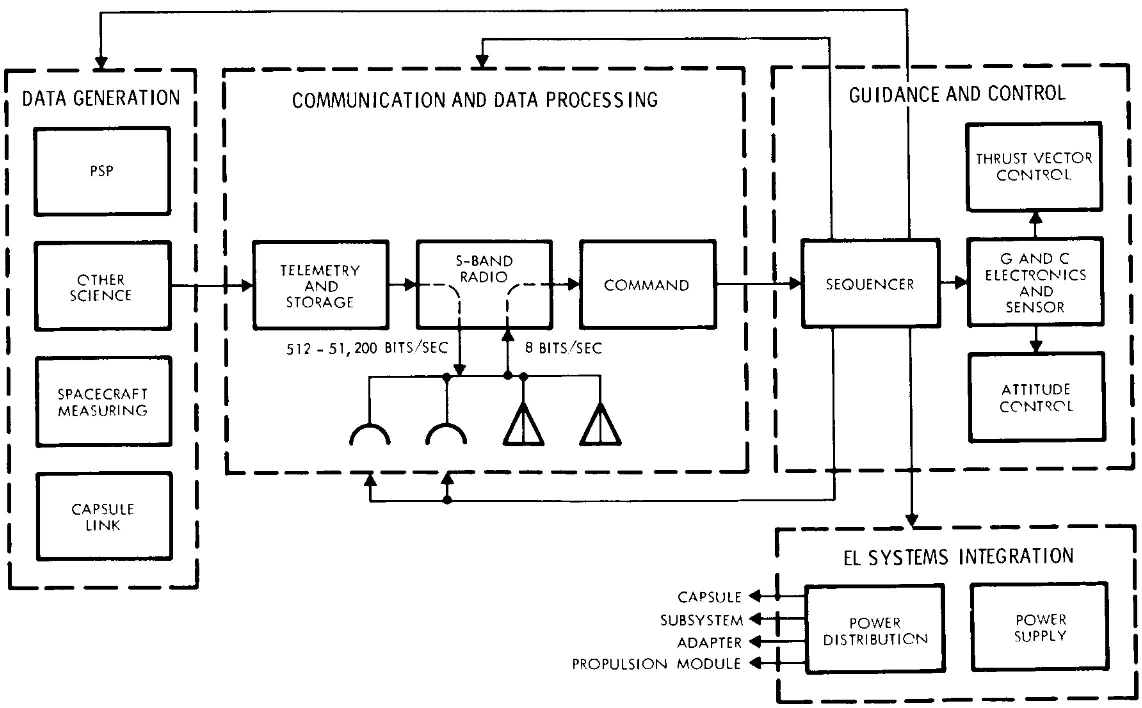


Figure 2

DATA GATHERED BY THE SPACECRAFT flows through the telemetry subsystem to the S-band radio subsystem for transmission at rates varying between 512 bits/sec and 51.2 kilobits/sec according to range and type of data. Control of the spacecraft is centered in the sequencer, which can be modified by commands from the earth transmitted at 8 bits/sec.

c) Electrical Systems Integration Group

- The Electric Power Subsystem contains the solar power supply, the batteries, the power control, DC/DC conversion, and the master clock.
- The Electrical Distribution and Pyrotechnics Subsystem routes power and signals to the electrical subsystems, the propulsion module, and the adapter; it does all power switching; it includes the firing circuitry of the pyrotechnics.

TRW has selected an overall spacecraft configuration with complete modularity. The equipment module houses all electrical subsystems and will be assembled and tested before mating with the propulsion module.

In the design of each of the subsystems, emphasis has been on achieving best performance consistent with a high confidence of reliable implementation for the 1973 mission, and minimum modifications for possible upgrading for later missions.

The recommended communication system uses a (32, 6) biorthogonal block code and is capable of transmitting at the rate of 51.2 kilobits/sec during the first three weeks in orbit about Mars. This is based on a conservative 6 db threshold sensitivity, a nominal trajectory (190×10^6 km at encounter), a 9.5-foot antenna on the spacecraft and the 210-foot, 45°K earth receiving station. The (32, 6) block code more than doubles the link capacity for the same effective radiated power. It has been selected over sequential coding because of its proven performance and real state of hardware development. This transmission capability enables an initial reconnaissance of large areas of Mars for selecting the capsule landing site. Mapping for scientific purposes continues thereafter at reduced transmission rates. The average transmission rate over the first six months is 18 kilobits/sec (for nominal trajectory with 190×10^6 km at encounter). From the low-gain antenna, the communication link can just support emergency telemetry at 8 bits/sec out to the latest arrival distance (240×10^6 km).

The recommended planetary mapping system is a film system for medium and high resolution plus color TV for coarse resolution (see Volume 5 and Volume 11). Nevertheless, the all-TV mapping system of the hypothetical science payload is reflected in the description of the telemetry and data storage subsystem in Section 1 of this volume since it presents the greatest demands on that subsystem. If the all-TV payload is designed to provide comparable performance to a film system, a severe tape recorder development problem arises, as discussed in Section 1 of this volume.

The handling of the engineering and nonvideo science data is not affected by the choice of photo-imaging system. Engineering data is transmitted directly at 512 bits/sec modulated on a 1 MHz subcarrier, or it is stored with the nonvideo science data on two interchangeable tape recorders with 7.5×10^7 bit capacity for each. The telemetry system is generally laid out as a centralized system with the exception of remote multiplexers on the planetary scan platform (in order to save wires going through the gimbal of the platform). It has excess capacity and a flexible format generator to accommodate the foreseeable growth and variations in payload.

The computer and sequencer exercises on-board control of the spacecraft with minimum intervention from the ground. These functions are also centralized with the exception of a remote decoder located on the platform (for the same reason as the multiplexers). It has the flexibility and sufficient capacity to control the type of experiment packages mentioned for possible upgraded versions of the spacecraft discussed in Volume 6.

The command subsystem is designed for 8 bits/sec; it requires a maximum of two minutes for acquisition.

The solar array is fixed (no deploying panels). For the recommended spacecraft of the 1973 mission, 226 ft^2 are used of a total available area of approximately 300 ft^2. The 226 ft^2 deliver 836 watts in Martian orbit at 1.62 AU. The maximum capacity would be 1099 watts at this distance. Three nickel-cadmium batteries are recommended with 16 amp hrs capacity each.

A cold gas (nitrogen) attitude control system is recommended for reasons of reliability. It has 12 thrusters operating at a thrust level of either 0.2 pound (low pressure) or 3 pounds (high pressure). The gas supply is sized to permit a thruster to fail open and yet achieve all mission objectives. The attitude limit cycles is ±0.5 degree. All control sensors are fully redundant: two Canopus sensors, two fine sun sensors, four coarse sun sensors, and two sets of inertial references (three gyros and one longitudinal accelerometer). The engine thrust vector pointing accuracy is 0.43 degree (3σ) or better.

The overall reliability estimate of these subsystems is 0.776 for a 6800-hour mission. This reliability has been achieved through careful design and implementation of redundancy at the component and black box level. This represents a somewhat conservative estimate inasmuch as it did not account for alternate and degraded modes of operation as is discussed in the detailed reliability calculations presented in Volume 2, Section 3.8, and Volume 6, Section 3.7.

The subsequent descriptions of the subsystems are supplemented by supporting analysis and tradeoff studies reported in the appendices. One study with potentially far-reaching implications on mission flexibility and for maximizing the return of meaningful data is the use of a centralized, general purpose computer. Since we have taken only the first step towards studying this concept, namely, using the computer for the control subsystem (see Appendix N), no definitive conclusion can yet be reached.

The most critical development item is the TWT and also, if an all-TV mapping system would be specified, the data storage device.

1. TELEMETRY AND DATA STORAGE

1.1 SUMMARY

The telemetry and data storage subsystem takes data from the science subsystem, the capsule, and the diagnostic measuring circuits (engineering data) and converts it into a form suitable for transmission by the digital S-band communications system. The engineering and science data can be transmitted real-time or can be stored digitally on tape recorders for playback at a later time. Tape storage is used as a buffer whenever data is being generated faster than the communications link can accept it. Tape storage of engineering data is also used during maneuvers as a backup in case of interruption of communications. During capsule descent, the formated TV and engineering data is received via the RF link and either stored or directly transmitted to earth. Programmable data format capability and seven data transmission rates are available for the recommended spacecraft. This makes it possible to meet the varying data requirements of different phases of the mission and of different missions without redesign of hardware. The maximum data transmission rate is 51.2 kb/sec transmitted from the high gain 9.5 foot parabolic antenna. An emergency telemetry rate of 8 bits/sec transmitted from the omnidirectional antennas enables diagnostic engineering information to be retrieved from the spacecraft under adverse circumstances. Mission elapsed time is transmitted within all data formats to correlate sampled data to mission phases.

A versatile centralized telemetry data handling unit is supplemented by two remote multiplexers on the planetary scan platform (PSP), one for the PSP-mounted experiments and the other for the all-TV mapping system. The purpose of these remote multiplexers is to reduce the number of wires going through the PSP gimbal. Flexibility is designed into the central unit so that it can handle changing science, as well as changing engineering data requirements in the 1973 and succeeding missions without redesign. This is particularly valuable for the first Voyager mission in that it enables reprogramming of diagnostic engineering formats to meet unexpected conditions.

Before data emerges from the telemetry and data storage subsystem, it is coded into a biorthogonal code. This effectively raises the overall transmission gain margin of the communications link by about 3 db at the expense of increasing the transmission bandwidth. This can be accommodated within the frequency spectrum of 10 MHz allocated to the DSIF for the mission.

There are two interchangeable tape recorders for engineering and nonvideo science data in the recommended spacecraft; each can store 7.5×10^7 bits. The input tape recorder data rate is 12.8 kb/sec for the science and 512 bits/sec for the engineering. The output data from either tape recorder modulates the S-band carrier.

Engineering data is sampled from the spacecraft subsystems at a rate of 512 bits/sec, and is normally transmitted directly. However engineering data is also stored during maneuvers for later playback.

The telemetry and data storage subsystem described here contains four additional tape recorders capable of storing data from the hypothetical all-TV mapping subsystem. These recorders accept data at the rate of 697 kb/sec or at other prevailing bit rates. This data also modulates the S-band main carrier. Each of these recorders stores up to 9×10^8 bits and represents a substantial development effort. As discussed in Volumes 5 and 11, the recommended science payload uses an Eastman Kodak film camera. The use of this payload removes the requirement for the high performance tape recorder associated with the hypothetical all TV payload. Nevertheless, the all-TV mapping system is reflected in the telemetry and data storage subsystem description of this volume since that system presents the greatest demands to the subsystem.

The location on the spacecraft of the telemetry and data storage subsystem for the all-TV system is shown in Figure 1-1. A preliminary specification and a block diagram of the subsystem is given in Figure 1-2.

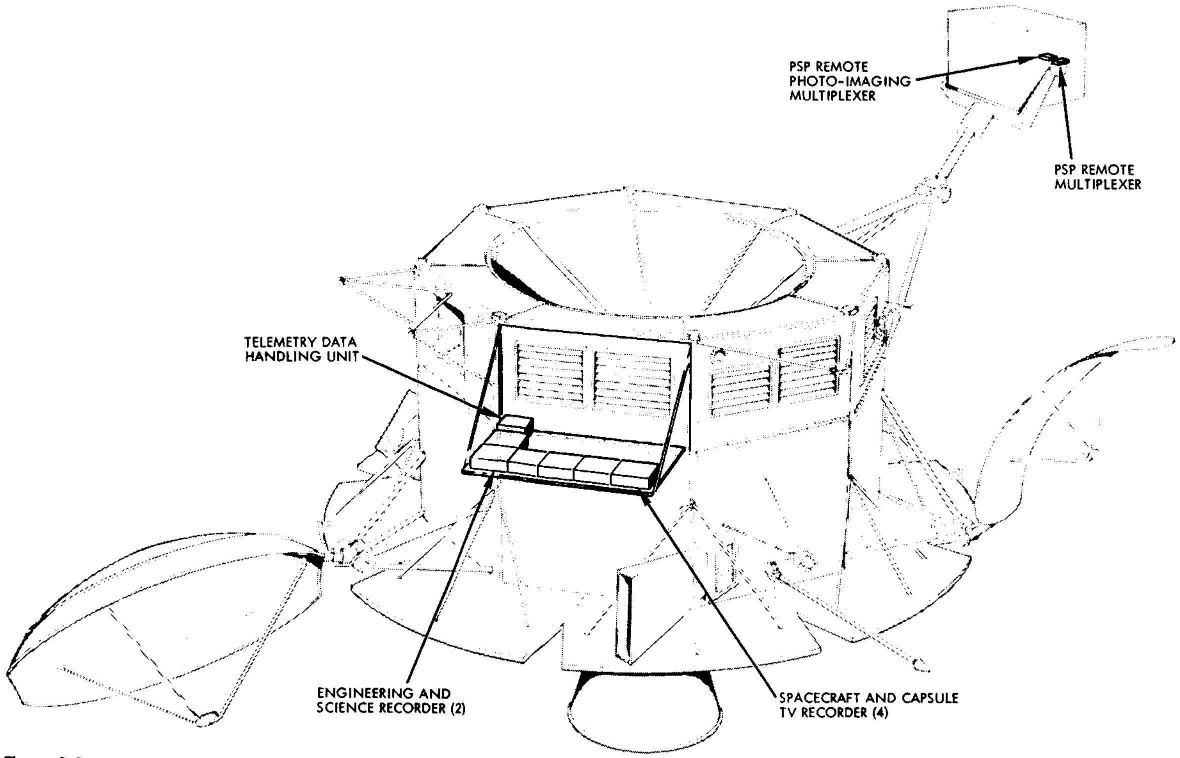


Figure 1-1

THE TELEMETRY AND DATA STORAGE SUBSYSTEM has tape recorders and the central telemetry data handling unit mounted together on one panel of the equipment module. A remote photo-imaging multiplexer serves the photo-imaging equipment on the planetary scan platform. A second remote multiplexer serves other science instruments on the platform.

1.2 SUBSYSTEM FUNCTIONAL DESCRIPTION AND PERFORMANCE

All data except that from the television subsystem is ordered into a serial time sequence by data multiplexers under the direct control of a flexible format generator (Figure 1-3). Preprogrammed formats can be selected by ground command or new formats can be programmed into the format generator from the ground. Television video data is handled by five independent video data encoders. Analog and digital signals are formatted into 6-bit words for engineering and video and 8-bit words for science data. Six discrete on-off signals are combined into one word, either in the multiplexer or within the source. In the latter case, the 6-bit word containing stored binary data is shifted out of the source register into the multiplexer as a digital word.

The functional block diagram, Figure 1-3, of the telemetry and data storage subsystem illustrates the required functions for an all-TV mapping system. The video engineering data and frame sync is combined with the video data into one serial digital output data stream.

PRELIMINARY SPECIFICATION

Telemetry and Data Storage Subsystem

Purpose

Formats and codes data from the science payload and measuring points in the spacecraft for digital transmission by the S-band communications system. Store this data during critical operations or when data rate is higher than communication link can handle.

Performance Characteristics

BIT RATE

(Main Carrier)

Split phase	51.2 kb/sec
	25.6 kb/sec
	12.8 kb/sec
	6.4 kb/sec
	3.2 kb/sec

(Subcarrier)

Normal	512 b/sec
Emergency	8 b/sec

ENCODING ACCURACY

Science	8 bits
Engineering	6 bits
Video	6 bits

DATA FORMATS

Programmable for engineering and science

Fixed frame size for video

TYPES OF DATA INPUT

Analog (0 to 5 volts)
Discrete
Digitals

STORAGE

Video data recorders 3.6×10^9 bits (4 recorders)

Engineering and science recorders 1.5×10^8 bits (2 recorders)

OUTPUT

Biorthogonal block code
Modulation
Biphase (Subcarrier)
Split phase (Main Carrier)

SIZE, WEIGHT AND POWER (EXCLUDING TAPE RECORDERS)

Central telemetry data handling unit

Size	480 in.^{3} (10 x 8 x 6 in.)
Weight	11 lb
Power	6 watts

Remote science multiplexer (located on PSP)

Size	20 in.^{3} (4 x 5 x 1 in.)
Weight	2 lb
Power	500 mw

Remote photo-imaging multiplexer (located on PSP)

Size	40 in.^{3} (4 x 5 x 2 in.)
Weight	2 lb
Power	2 watts

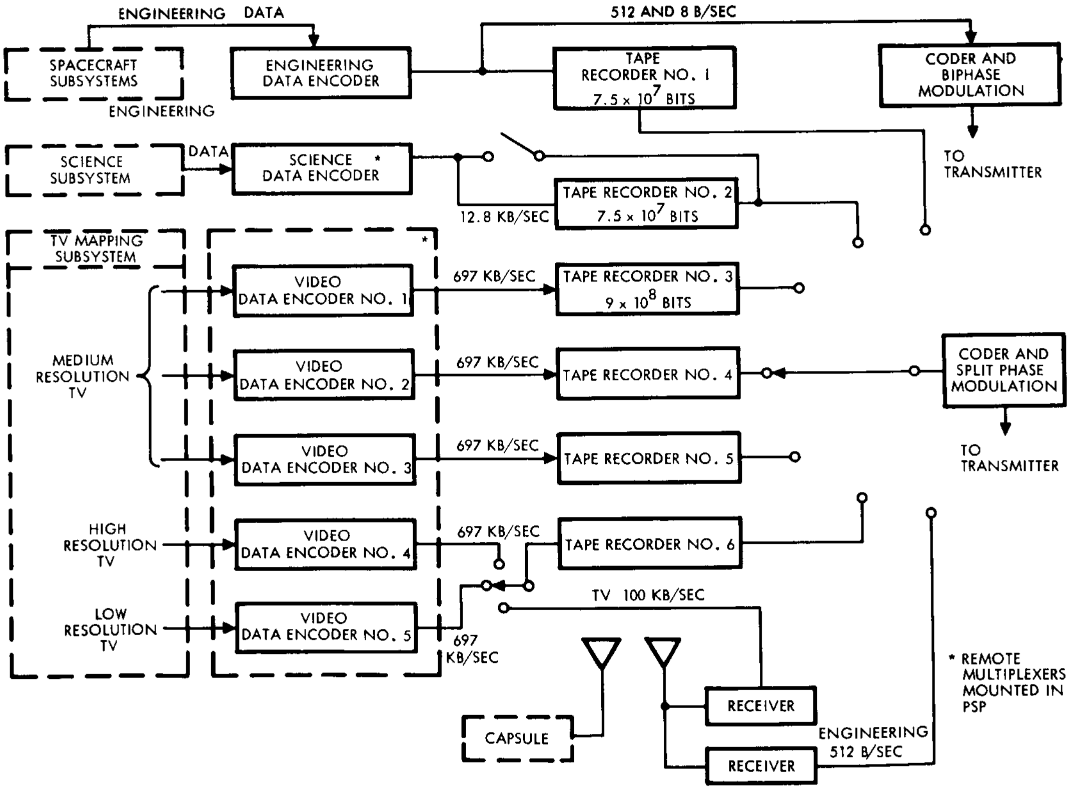


Figure 1-2

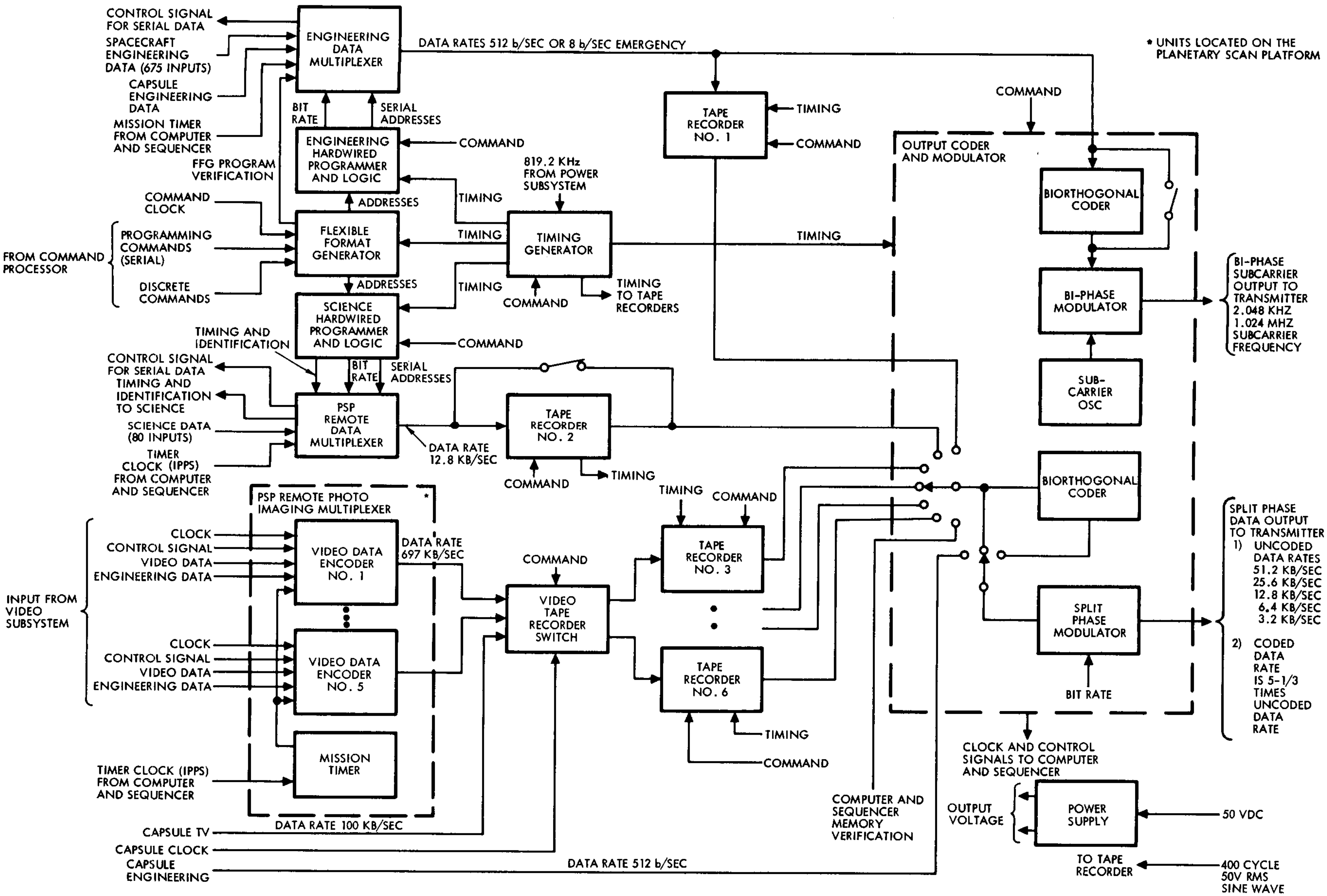


Figure 1-3

TELEMETRY AND DATA HANDLING SUBSYSTEM accepts engineering data, experiment science data and video data and formats these data into appropriate formats for telemetry transmission. Engineering data and experiment science data multiplexing and formatting is under the control of a programmable flexible format generator; however, the video data is sampled and formatted by five independent video data encoders. Tape recorders store sampled data for transmission at the prevailing bit rate. The capsule data is preformatted by the source and is either stored or transmitted directly.

Formatting flexibility is gained through the use of the programmable flexible format generator while modularization of the data multiplexer assemblies makes possible easy upgrading for later missions. The flexible format generator is backed up with a hard-wired programmer capable of providing one fixed data format for each multiplexer. The hardwired programmers are brought into operation by ground command.

The flexible format generator provides serial addresses to the multiplexer. The address bits are decoded by the address decoder and driver within the multiplexers which in turn selects the desired data input. The flexible format generator provides different address information to the engineering data multiplexer and to the remote multiplexer on the PSP at different address rates. For example, if the engineering multiplexer data rate is 512 bits/sec and the PSP remote multiplexer data rate is 12.8 kb/sec, then 25 addresses are provided to the second multiplexer for every one provided to the first.

The FFG provides an efficient means to achieve format flexibility during the mission as well as mission-to-mission. Flexibility during the mission is achieved by selecting various stored formatting programs by a single ground command to meet the formatting requirements during various mission phases. Flexibility also is achieved by the ability to fill new data formatting programs into the FFG via ground commands which provides a means to efficiently deal with unanticipated data requirements which could develop during the mission.

Since each data input is addressed by a random access FFG word, each data input can be sampled at any desired sequence and rate within the constraints of the programmed format. For example, if an input is required to be sampled at a rate equivalent to three times the main frame rate, then, the same word may be sampled at the beginning, middle and end of the frame. This is not possible with fixed programmers.

The FFG is capable of generating various main and subframe lengths. For example, the baseline science format requires a 32 word main frame whereas the upgraded science format requires a 64 word main frame. These two frame lengths are readily achievable by the FFG.

Analog data from the multiplexer is converted into digital words by an analog-to-digital converter and a composite serial bit stream of data, frame synchronization signals, identifications, etc. is developed by each data multiplexer unit. The main stream of engineering data then passes through a biorthogonal coder and a biphase modulator and into the S-band communications system for transmission. The subcarrier frequency is 1.024 MHz. This data stream may also be recorded on one of two tape recorders for playback later in the mission via a biorthogonal coder and splitphase modulator and then to the main carrier S-band transmitter. The main carrier can also accept direct or recorded data from the science instruments on the PSP or recorded video data. The biorthogonal coders can be switched out when desired.

A timing generator circuit is provided to generate the required clock frequencies from the spacecraft master clock (819.2 kHz from the power subsystem). Synchronized clock pulses are generated for the main and subcarrier data processing. Clock pulses for the biorthogonal coders are generated at 5-1/3 times the prevailing bit rate frequency.

A data output gate is provided to select the data sources to be transmitted via the main carrier.

The telemetry and data storage system for the hypothetical all-TV mapping system contains four tape recorders of identical configuration. The recorder assigned to the high and low resolution television cameras can be substituted for a medium resolution camera recorder, providing backup redundancy. Alternatively, a medium resolution recorder can be used in place of the high and low resolution recorder, providing another redundancy mode. Each recorder has separate power, control and signal lines.

The spacecraft engineering and science data recorders are identical units and can be switched to record from either source, thus providing redundancy of operation in case of failure of one unit.

The data storage subsystem has a total storage capacity of 51×10^8 bits, as follows:

- Medium resolution television data recorders: three recorders each with a capacity of 9×10^8 bits

- High or low resolution television data recorder: one recorder with a capacity of 9×10^8 bits
- Spacecraft engineering data recorder: one recorder with a capacity of 7.5×10^7 bits
- Science data recorder: one recorder with a capacity of 7.5×10^7 bits

Track switching is used on all recorders to reduce the physical length of magnetic tape required by recording on the top half of the tape in the forward direction and the bottom half of the tape in the reverse direction.

The telemetry data rates provided by the telemetry and data storage system have been optimized by means of tradeoff studies that take account of the mission data requirements and S-band communications constraints. The main constraint is on the maximum data rate available for television mapping data, and the reduction of this rate as the mission proceeds and spacecraft range increases. Within this constraint, the highest bit rate was selected such that it would last for approximately 2/3 of the first month in order to conduct the reconnaissance for the capsule landing site.

The switchable bit rates selected as a result of these studies are as follows:

Main Carrier (kb/sec)	Subcarrier (bits/sec)
51.2	512
25.6	8 (emergency)
12.8	
6.4	
3.2 (emergency)	

The telemetry system is a synchronous two-channel PCM system. The subcarrier data output is biphase modulated whereas the main carrier output is splitphase modulated. All transmission frames contain a 24-bit pseudorandom code at the beginning of each data frame to enable rapid frame synchronization in the decommutator. A word is provided following the frame sync code to identify the subcommutator or subframe. An additional word is provided to identify the current operating mode and bit rate.

Six main carrier transmission formats are used on the S-band main carrier:

- Main carrier format No. 1, playback of stored science data
- Main carrier format No. 2, playback of stored engineering data recorded during maneuvers
- Main carrier format No. 3, real time transmission of science data
- Main carrier format No. 4, transmission of computer and sequencer memory contents
- Main carrier format No. 5, playback of stored video data
- Main carrier format No. 6, relay of capsule engineering data during capsule descent.

Two basic data transmission formats are used on the S-band subcarrier:

- Subcarrier format No. 1, engineering data only, includes high rate propulsion, guidance and control data. This mode is used primarily during maneuvers and for failure analysis.
- Subcarrier format No. 2, is a combination of spacecraft and capsule engineering data and low rate science data such as field and particle measurement data (upgraded spacecraft only). This mode is used during cruise.

The basic telemetry and data storage subsystem interface is shown in Figure 1-4.

1.2.1 Engineering Data Flow

In the recommended spacecraft, engineering data is transmitted in two basic modes. Mode 1 is used during cruise, and Mode 2 in premaneuver phases and during maneuvers. In both modes the engineering data is transmitted continuously on a subcarrier at 512 bits/sec. Each word contains 6 bits.

TRW SYSTEMS

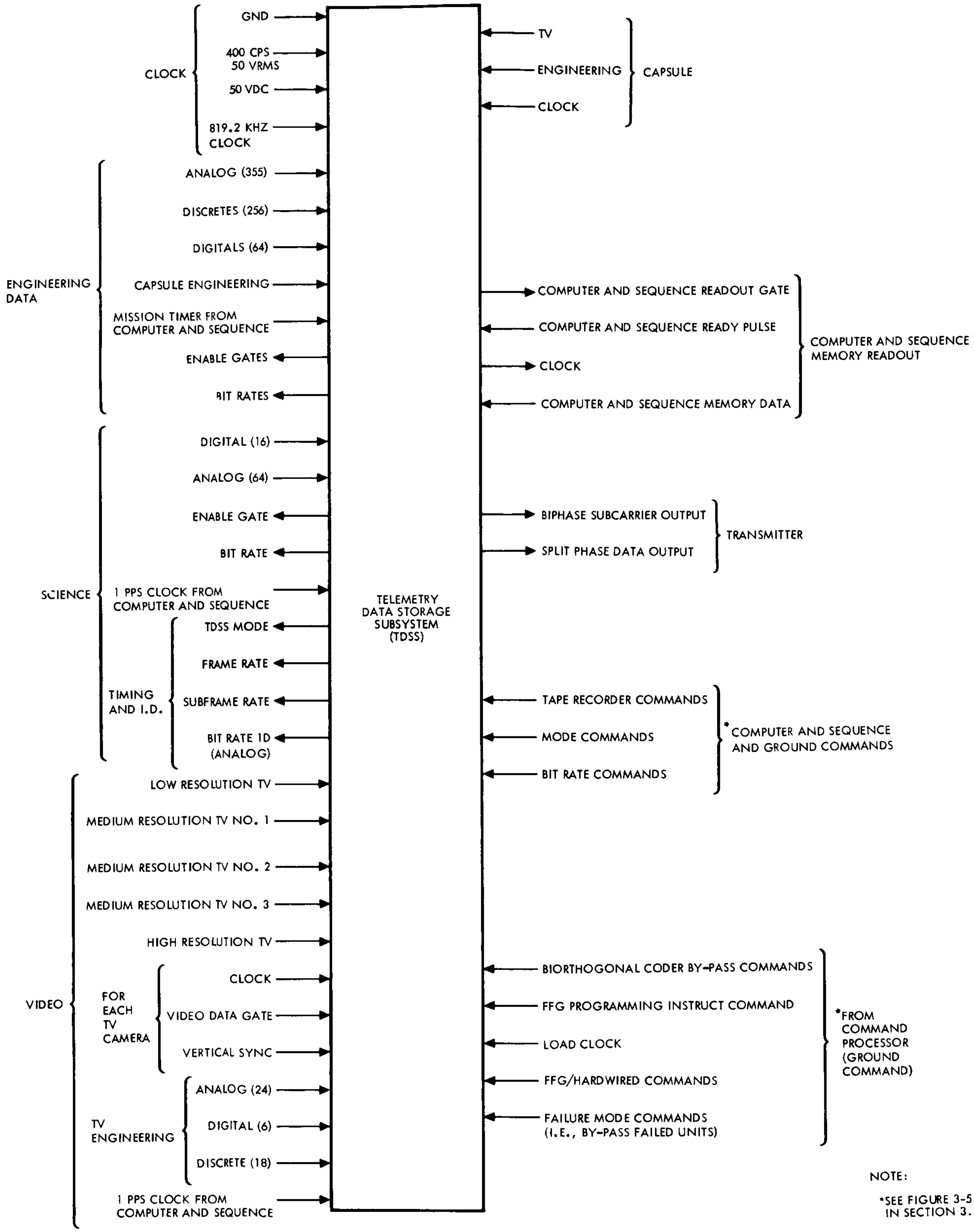


Figure 1-4
TELEMETRY AND DATA STORAGE SUBSYSTEM (TDSS) SIGNAL INTERFACE illustrates the basic interfaces between the TDSS and other subsystems.

In Mode 1 the nominal engineering main frame consists of 64 words:

- Six frame sync words and identification words
- Five subcommutators, each with 64 assignable words
- Fifty-three assignable words in the main frame.

In Mode 2 the nominal engineering main frame consists of 64 words:

- Six frame sync words and identification words
- Three subcommutators, each with 64 assignable words
- Fifty-five assignable words within the main frame.

Binary words greater than 6 bits in length are sampled by the multiplexer as multiple words. For example, a 20-bit word would occupy four 6-bit words.

The telemetry and data handling subsystem elements required to process engineering data is indicated by the shaded portions of Figure 1-5. The engineering data formats available are specified in Table 1-1.

Figure 1-6 illustrates a typical FFG format consisting of 64 6-bit main frame words and a number of subframes of various sizes illustrating the capability of the FFG to generate the required engineering formats.

Engineering data to be transmitted falls into the following four categories:

- Measurements required for the performance of flight operations. These include measurements for diagnostic purposes to select commanded alternate modes of operations.
- Measurements required to verify the performance of specific spacecraft and subsystem functions.

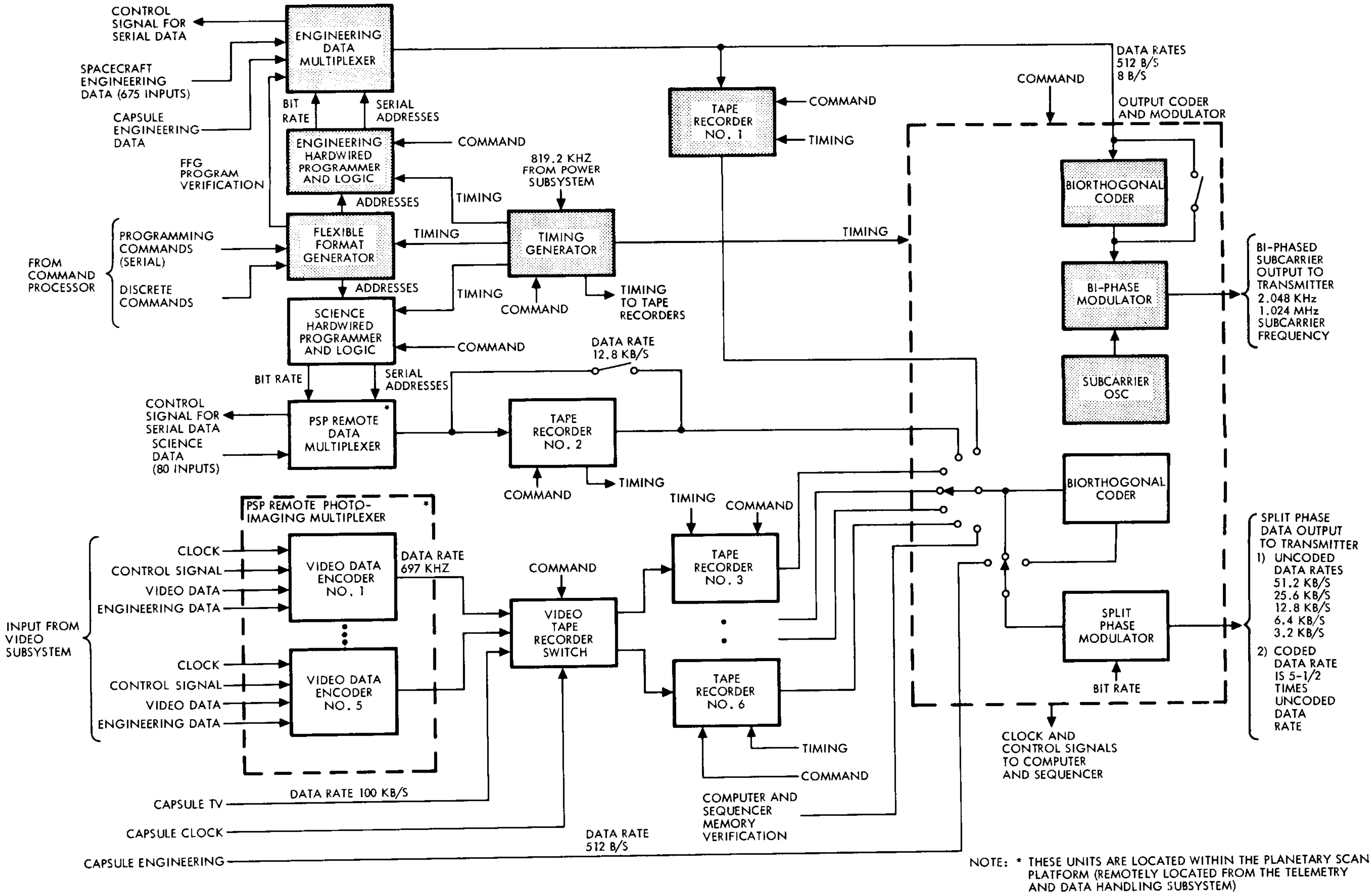


Figure 1-5
THE ENGINEERING DATA FLOW is identified by shading the blocks that are required to process this data.

Table 1-1. Engineering Data

Word size - 6 bits

Types of inputs

Analog — 0 to +5 volts

Discrete — 0 = False

+5 volts = True

Digital — 0 = False

+5 volts = True

Analog encoding accuracy — 6 bits

Frame size — programmable, typically 64 words

Subframe size — programmable, 8, 16, 32 and 64 words typical

Number of subframes — programmable

Frame rate (64 words at 512 bits/sec) — 0.75 sec/frame

Subframe rates at 512 bits/sec

64 words — 48 sec/frame

32 words — 24 sec/frame

16 words — 12 sec/frame

8 words — 6 sec/frame

Number of data channels - variable by increasing multiplexer gates up to a total of 1024 inputs

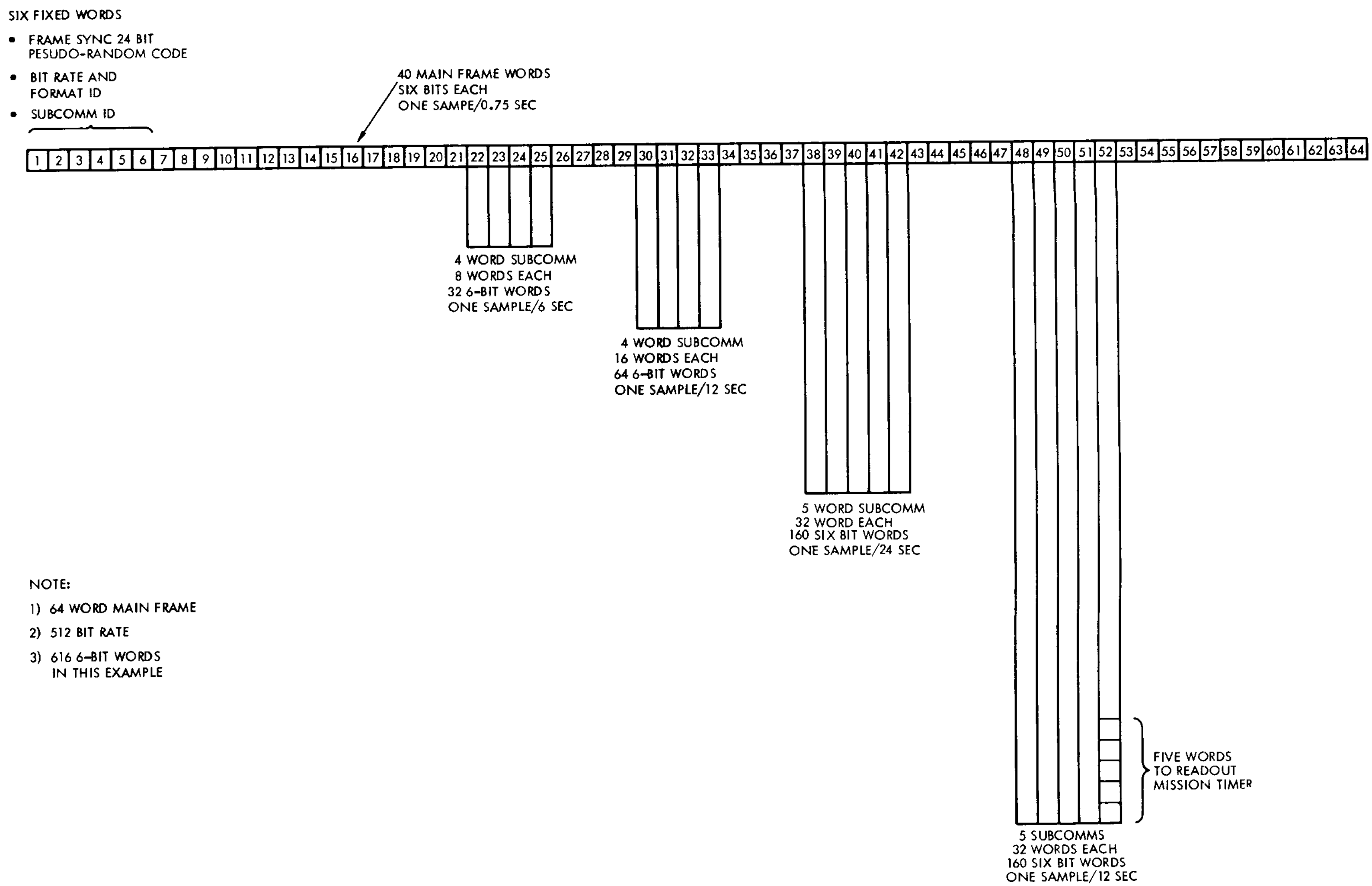


Figure 1-6
TYPICAL FFG FORMAT shown consists of 64 six bit mainframe words and a number of subframes of various sizes illustrating the capability of the FFG to generate the required format.

- Measurements to indicate the effect of the space environment on the spacecraft performance.
- Data to evaluate critical components; in particular, those newly developed for Voyager.

A summary of engineering data channels assigned for each spacecraft subsystem is shown in Table 1-2. A detailed list is given at the end of this Section.

Table 1-2. Telemetry Channel Requirements

	Mode 1		Mode 2	
	Main Frame	Subcom	Main Frame	Subcom
Engineering				
Guidance and control	24	40	34	40
Command and sequencing	4	36	4	22
Communications	4	67	-	1
Telemetry and data storage	-	28	-	-
Thermal control	-	40	-	-
Capsule	-	6	-	-
Power	-	48	-	15
Propulsion	8	46	16	46
Electrical distribution	1	13	1	8
Structure	-	2	-	-
Total	41	326	55	132
Science (Total)	17	34	-	-

Available Spare Channels for Selected Frame Size

	Mode 1		Mode 2	
	Main Frame	Subcom	Main Frame	Subcom
Engineering	11	59	0	62
Science	7	13	-	-

1.2.2 Science Data Flow

Science data, other than video data, is transmitted on the main S-band carrier at 12.8 kb/sec for real-time operation. It can also be stored on the science tape recorder and replayed at high bit rates when the prevailing bit rate is higher than 12.8 bits/sec. The science data format is programmable by the flexible format generator and is similar to the nominal format for the engineering data. However, the science format is only required to contain a maximum of 160 words. This includes the data requirements for the upgraded spacecraft discussed in Volume 6. The telemetry and data storage subsystem elements required to process spacecraft science data is shown by the shaded portions of Figure 1-7. Specifications for the available science formats appear in Table 1-3. A 64-word main frame format is illustrated which reflect the upgraded data format requirements. The hypothetical science payload requires only a 32-word main frame.

The telemetry lists at the end of the section list the channel requirements expected for different instruments in the hypothetical science payload.

1.2.3 Video Data Flow

The high rate data emerging from the television mapping system is put into a fixed format by a remote photo-imaging multiplexer on the PSP which contains one video data encoder for each TV camera. This multiplexer is not under the control of the flexible format generator because the video data must be encoded and synchronized by the video data clock from each TV camera. It is located on the PSP instead of the planetary equipment panel (see Figure 1-1) in order to minimize the number of wires going through the PSP gimbal; for the television photo-imaging system, a horizontal scan line, approximately 18,000 bits, for the medium resolution TV, which constitutes a telemetry frame. This data contains all the required synchronizing and identification pulses to reconstruct one line of the picture on the ground. In addition, the remote photo-imaging multiplexer enters telemetry-frame

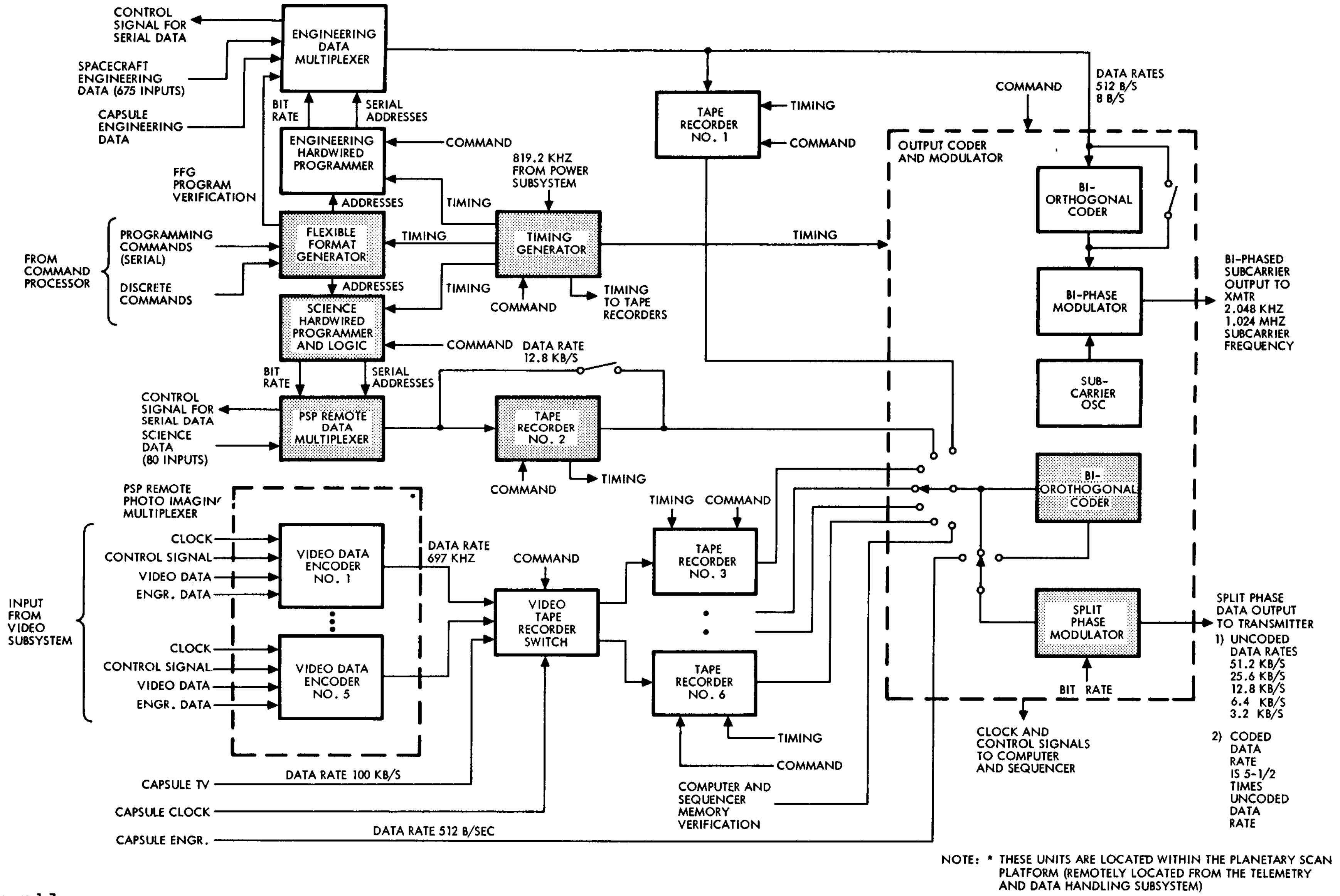


Figure 1-7
THE NONVIDEO SCIENCE DATA FLOW is identified in this block diagram by the shaded blocks.

Table 1-3. Science Data

Word size - 8 bits

Types of Inputs

Analog: 0 to + 5 volts

Digital: 0 = False

+ 5 volts = True

Analog Encoding Accuracy - 8 bits

Frame Size - programmable, typically 64 words

Subframe Size - 8, 16, 32 and 64 typical

Number of Subframes - programmable

Frame rate (64 words at 12.8 kb/sec) - 0.04 sec/frame

Subframe rate, typical 16 word subframe - 1.56 sec/frame

Number of Channels - variable; maximum 160

sync code bits at the start of each 18,000 video data horizontal scan line. At the end of a picture, approximately 4280 horizontal scan lines of data, the remote multiplexer will enter 22 TV camera engineering measurements. Sampling and encoding of video data is performed at the video data rate of 697 kb/sec with continuous uninterrupted sampling during each horizontal scan line. A separate video data encoder is provided for each camera, since some cameras are used simultaneously.

The telemetry and data handling subsystem elements required to process video data is indicated by the shaded portions of Figure 1-8.

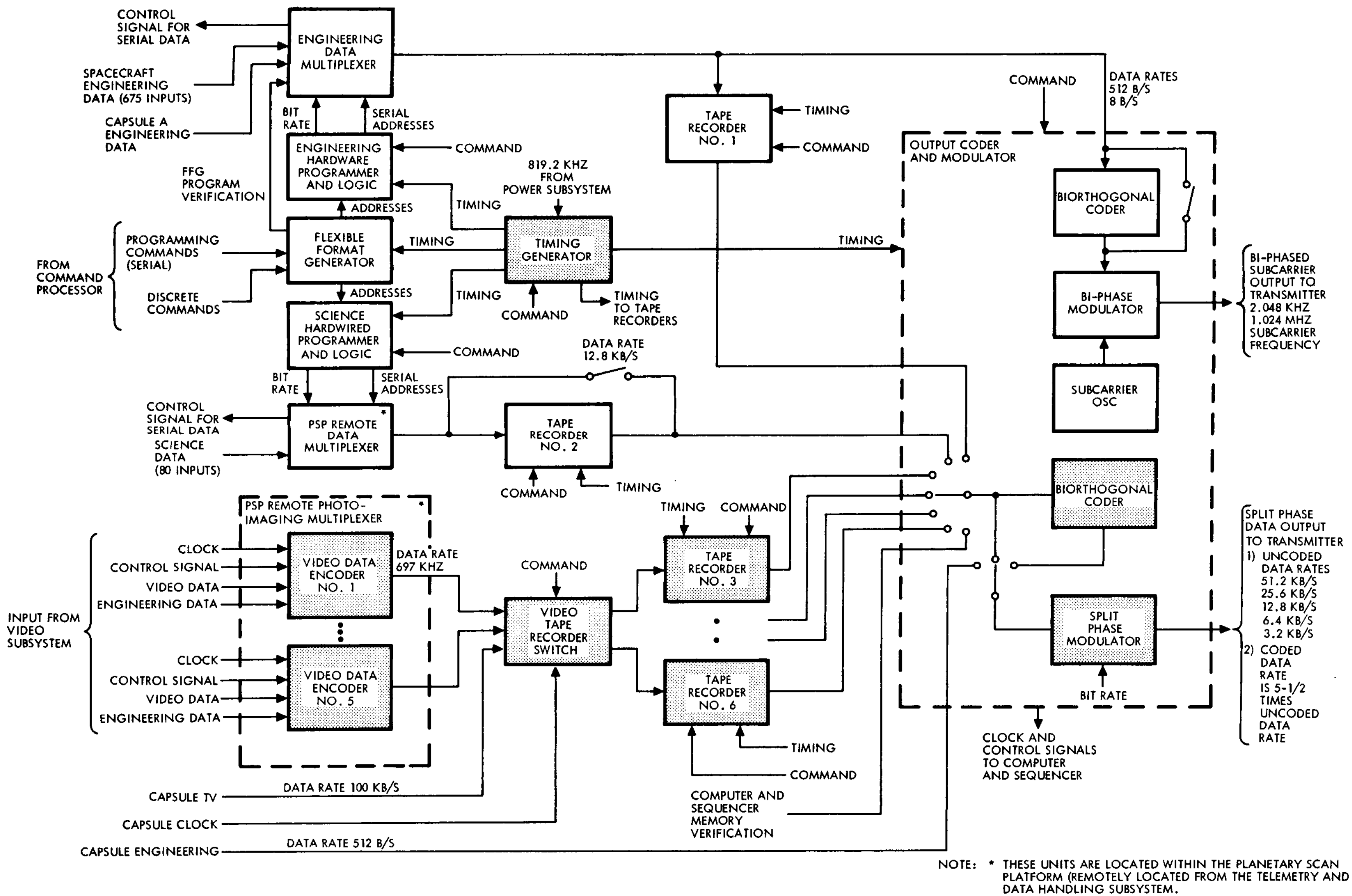


Figure 1-8
PROCESSING OF SPACECRAFT VIDEO DATA elements required to process spacecraft video data are shown shaded in the figure.

1.2.4 Capsule Data Flow

The telemetry capsule interface permits the spacecraft to transmit capsule data during launch through capsule-spacecraft separation, and to relay capsule data to earth subsequent to separation. The data flow takes three forms:

- Hard Wire

  Capsule data and clock pulses, prior to separation, are received by the telemetry at a low rate (≈ 10 bits/sec) via a hard line to the capsule. The capsule data is in serial digital form, asynchronous with the spacecraft telemetry clock. The telemetry and data storage subsystem buffers the data for inclusion in the telemetry format.

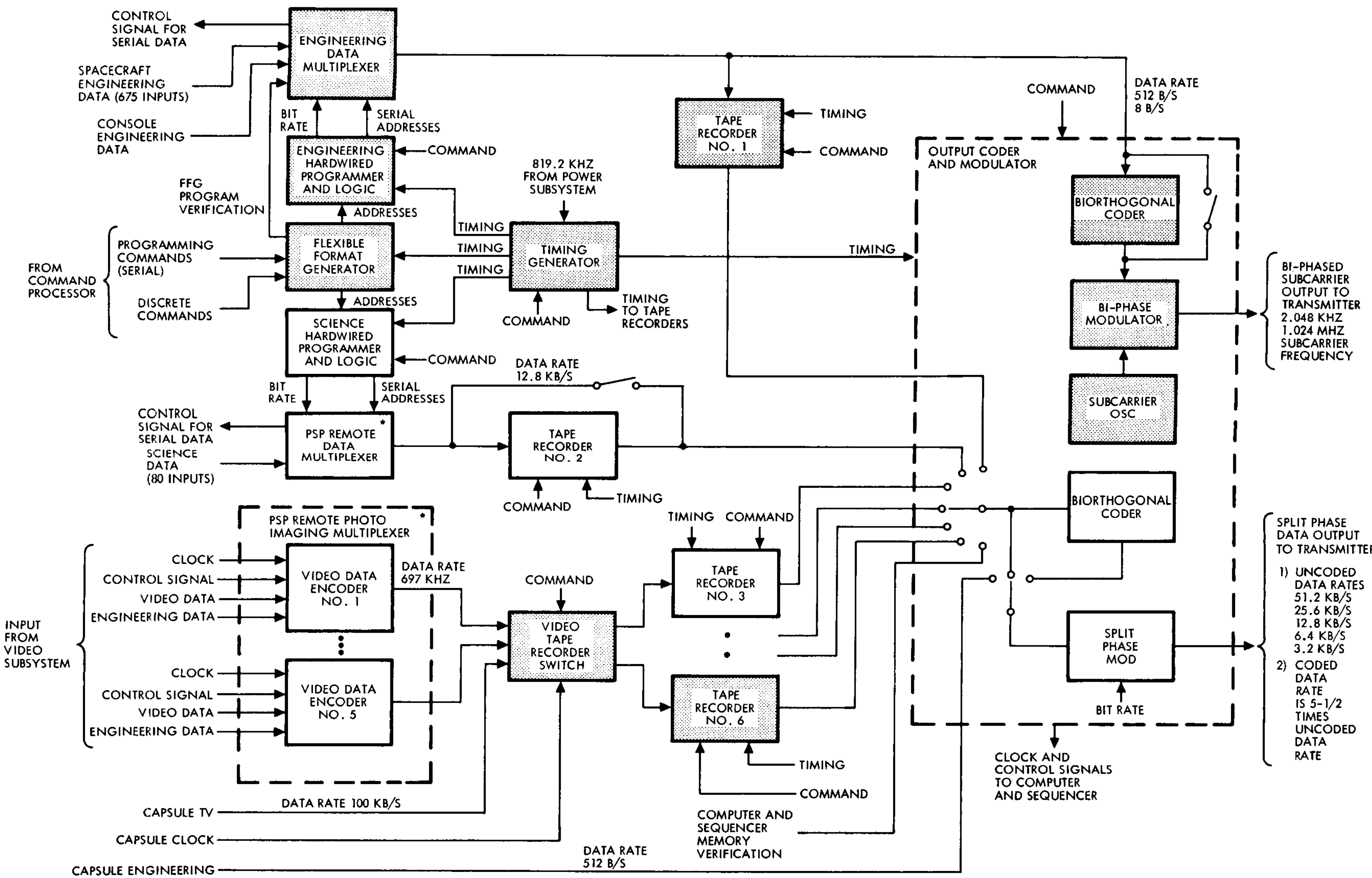
- RF Link

  Capsule data after capsule separation is received by the telemetry and data storage subsystem at rates of 100 kb/sec and 512 bits/sec via two RF links. Demodulated engineering data (512 bits/sec) is supplied to the subsystem asynchronous with the spacecraft telemetry bit rate by the capsule low-rate data demodulator. The 512 bit/sec data is accepted by the capsule buffer and formated into the spacecraft telemetry data stream. The high rate capsule data (video) is recorded on one of the spacecraft video tape recorders and played back at a slower rate, compatible with the telemetry bit rate. The recorder provides clock pulses to synchronize the data playback, and also provides an end-of-tape signal.

The telemetry and data storage subsystem processing of capsule data is indicated by the shaded portions of Figure 1-9.

1.2.5 Telemetry and Data Storage Subsystem for Recommended Science Payload

The telemetry and data storage subsystem for the recommended science payload discussed in Volume 5 (Figure 1-10) differs from the subsystem required for the hypothetical five-TV camera video subsystem. The recommended photo-imaging system consists of one film camera and one low resolution television camera. The data handling requirements are thus simplified. The four high capacity, high tape speed video



NOTE: * THESE UNITS ARE LOCATED WITHIN THE PLANETARY SCAN PLATFORM (REMOTELY LOCATED FROM THE TELEMETRY AND DATA HANDLING SUBSYSTEM

Figure 1-9
PROCESSING OF CAPSULE DATA. Elements required to process capsule data are shown shaded in the figure.

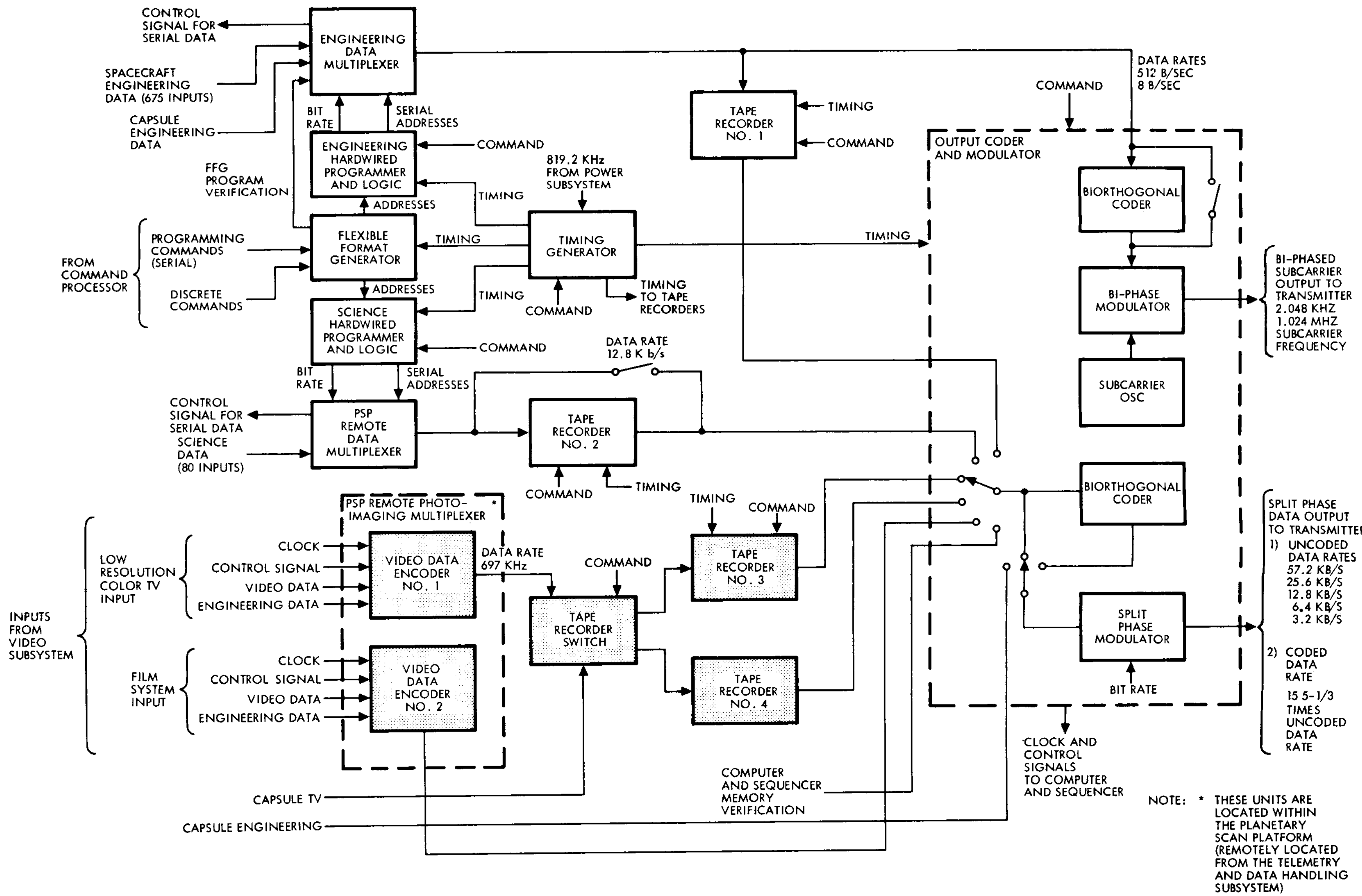


Figure 1-10
THE RECOMMENDED TELEMETRY AND DATA STORAGE SUBSYSTEM requires only four tape recorders, as the majority of the video data is stored on photographic film within the photo-imaging subsystem. The two video recorders that are required have to meet much less stringent requirements than those required for the hypothetical payload.

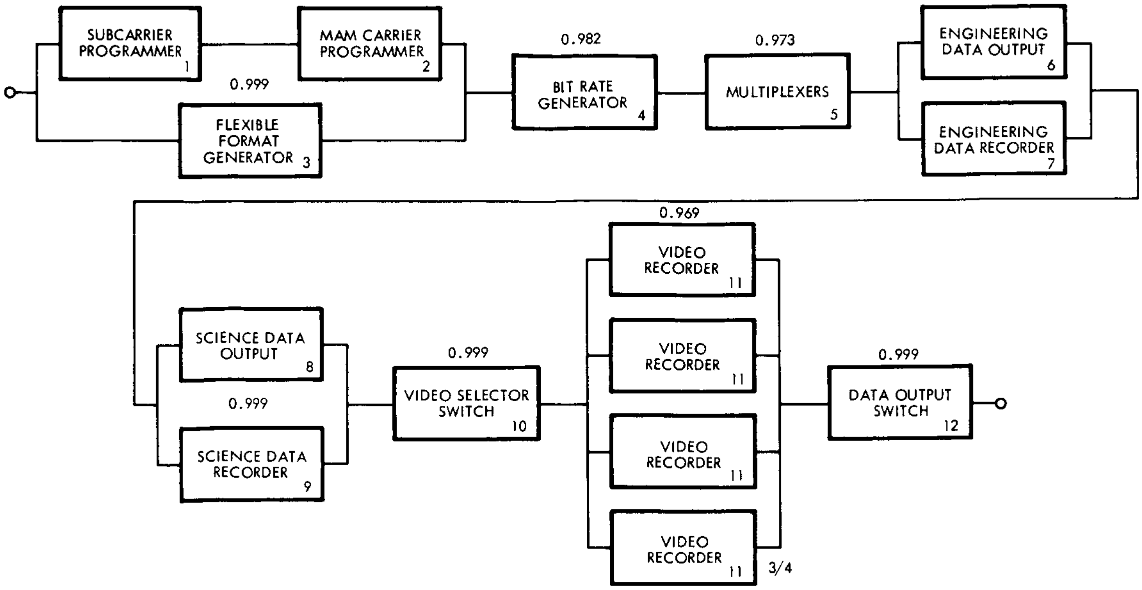
recorders are replaced by two lower speed, lower capacity tape recorders. In addition, the PSP remote photo-imaging multiplexer is simplified since only two of the five video data encoders are required for the recommended payload.

The recommended payload requires fewer video tape recorders because first, the film system does not require a tape recorder and secondly, the use of only one low resolution TV necessitates the use of one, less complex, tape recorder and one tape recorder for backup. Therefore, only two video recorders are required on the recommended spacecraft versus the four video recorders required for the hypothetical television payload. The video tape operational requirements are eased (tape speed reduced from 50 to 25 in./sec) since the video data rate can be slower for the low resolution color TV than for the hypothetical medium resolution TV systems (approximately one-half the rate). The net effect is the simplification of the PSP remote photo-imaging multiplexer and the elimination of two video tape recorders.

1.2.6 Failure Modes and Redundancy

The reliability block diagram for the telemetry and data storage subsystem used with the hypothetical science payload is shown on Figure 1-11. This subsystem incorporates considerable redundancy and alternate modes of operation. The hard wired programmers (1 and 2) act as a backup for the flexible format generator (3). If the flexible format generator fails, data can be sequenced in a given preprogrammed order rather than in variable order, the capability that exists when the flexible format generator is operating properly. Engineering data can be transmitted real-time (6) or can be stored (7) and transmitted when more convenient. Likewise, science data can be transmitted real-time (8) or can be stored (9). TV-Camera data must be stored and then retransmitted. Four recorders (11) are provided for this purpose and mission success is assured if only three of four recorders are operational.

In addition to the redundancy shown in Figure 1-11, the multiplexer (5) incorporates quad-redundant circuitry. The failure rate shown for this equipment is the effective component failure rate including this quad-redundancy.



$$R_{TDSS} = \left[R_1R_2 + R_3 - R_1R_2R_3\right]\left[R_4\right]\left[R_5\right]\left[R_6 + R_7 - R_6R_7\right]\left[R_8 + R_9 - R_8R_9\right]\left[R_{10}\right]\left[R_{11}^4 + 4R_{11}^3(1 - R_{11})\right]R_{12}$$

$$= 0.921$$

Figure 1-11

RELIABILITY OF THE TELEMETRY AND DATA STORAGE SUBSYSTEM is estimated to be 0.921 on basis of above block diagram. Detailed calculations are presented in Volume 2, Section 3.8.7.

Table 1-4 lists the components included in the subsystem and the reliability breakdown. The parts failure rates are listed in Volume 2 Appendix E.

1.3 COMPONENT DESCRIPTION

1.3.1 Central Telemetry Data Handling Unit

This unit physically includes the engineering data multiplexers, the flexible format generator, the hardwired programmers and logic, analog-to-digital converters (contained as part of the multiplexers), the biorthogonal coders, subcarrier oscillators, NRZ data modulators, the timing generator and the video tape recorder switching unit. The tape recorders and the remote multiplexer units are not part of the central telemetry data handling unit although they do interface with each other. The central data handling unit is the controlling center for all data handling functions within the spacecraft.

Table 1-4. Telemetry and Data Storage Subsystem - Equipment Reliability Breakdown

	Name	Failure Rate (λ) (bits/10^9)	Probability of Success $e^{-\lambda T}$ = 6800 hr
1.	Hard-wired programmer (subcarrier)	330	.9978
2.	Hard-wired programmer (main-carrier)	240	.9983
3.	Flexible format generator	6067	.9595
4.	Bit rate generator and sync. generators	2730	.9816
5.	Subcarrier and main carrier multiplexers	4035	.9730
6.	Engineering data direct output	910	.9938
7.	Engineering data tape recorder	12000	.9210
8.	Science data direct output	60	.9999
9.	Science data tape recorder	12000	.9210
10.	Video data tape selector	150	.9990
11.	Video data tape recorders	12000	.9210
12.	Data output coder and switch	150	.9990

1.3.1.1 Engineering Data Multiplexer

The data multiplexers connect each data source in sequence to an analog-to-digital converter or to a digital comparator. The recommended multiplexers use metal oxide semiconductor (MOS) switches since they are small, consume low power, are low in cost, and have been proven very effective in other TRW space programs.

The multiplexers have been designed such that the failure of any one gate will not affect more than four data channels. The multiplexers are protected against overvoltage by current limiting resistors on each data input gate. For each analog multiplexer:

- Leakage feedback currents to the ON source due to leakage currents of all OFF gates are kept to a minimum. This limits the maximum allowable source resistance and the highest operating temperature of the system.
- Crosstalk between the selected channel and all OFF channels is kept to a minimum due to finite impedance of OFF gates.
- Stabilization time for the multiplexed analog signal due to accumulated capacitance of all OFF gates and cabling and stray capacitances is minimized.
- Two or more levels of multiplexing are employed to avoid loss of all channels when failure of one analog gate occurs.

The various types of gates are shown in Figure 1-12 and were examined for this application. The two-diode gate is simple, easily driven, and has high-fault immunity, but is applicable only to low-accuracy systems. This is due to the relatively high-source feedback currents, gate attenuation, and mismatch. The four-diode bridge gate nominally has no feedback current to the source, but, like the two-diode gate, requires matched diodes and a floating drive. Additionally, the fault protection is difficult to achieve.

The transformer-driven Bright gate has negligible feedback current, low ON impedance, and offset, with fault protection equal to the transistor breakdown voltage. However, the transformer drive renders micro-miniaturization extremely difficult.

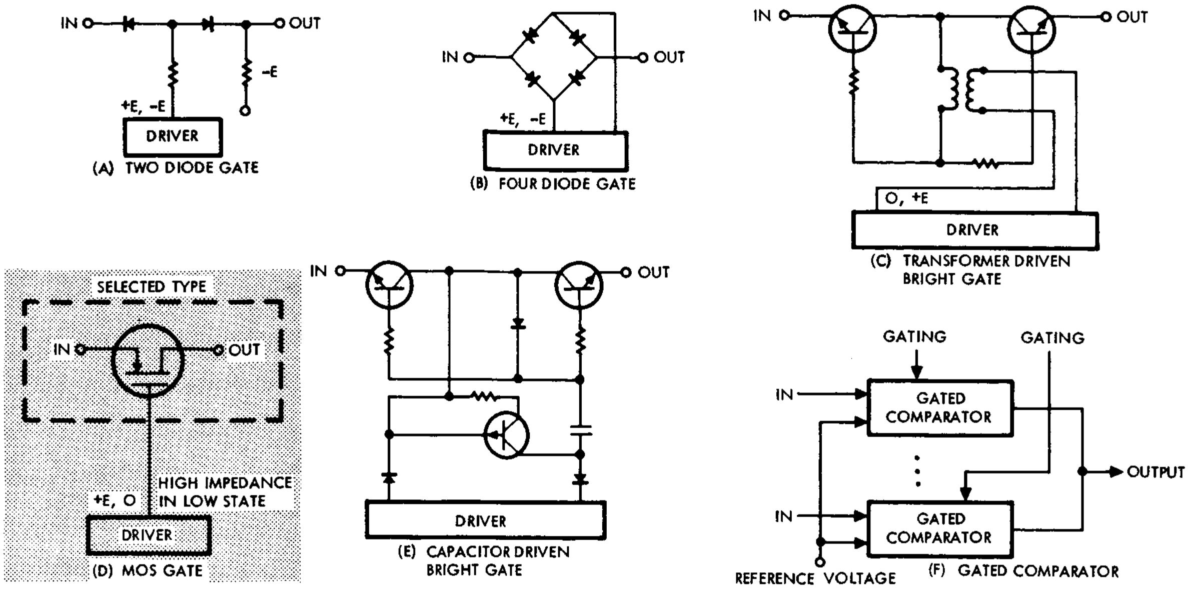


Figure 1-12
DIFFERENT TYPES OF GATES illustrate the techniques which have been used as multiplexer switches. The MOS gate (D) has been selected for Voyager.

The capacitor-driven Bright gate requires continuous drive power for each gate. Additionally, the gate driver is necessarily complicated because it must provide two polarities of drive to each gate. Also, at low-bit rates, the capacitor size becomes rather large, but this latter problem can be circumvented by converting at a high speed. However, pulse aperture techniques can be used which require additional complexity of sample-and-hold and other techniques. Lastly, the gated comparator technique is one in which a comparator is used for each analog input. Selection is performed by turning on the comparator. Although this technique could be used to fulfill the switching requirements, it is not recommended because of present high cost.

The switching device recommended is a monolithic MOS enhancement, P-channel quad gate, which is packaged within a 14-lead flatpack. Two configurations are used: four gates with a common drain output; and four independent gates. The entire multiplexer switching is performed with these two-gate configurations.

The basic gating arrangement recommended is shown in the Preliminary Specifications, Figure 1-13. Four levels of gating are used for both the analog and digital multiplexers. The first level consists

PRELIMINARY SPECIFICATION

Analog/Discrete/Digital Multiplexer

Performance Characteristics

Accomodates 64 digital inputs, 64 enable outputs, 256 discrete inputs and 384 analog inputs.

Expands to accomodate 128 digital inputs, 128 enable outputs, 384 discrete inputs and 512 analog inputs without major redesign.

Any one gate failure will not affect more than four channels, unless otherwise specified.

The multiplexer will not be permanently damaged by an input voltage of ± 35 volts, and will operate within specification after the removal of overvoltage.

An overvoltage of ± 16 volts on one input channel will not affect more than three other channels.

All channels will have an input impedance greater than 10 meghom when off and less than 2K ohm when on. This does not include the resistance of the input current limiting resistor.

The input impedance of any channel will be greater than 1 megohm when the multiplexer power is turned off.

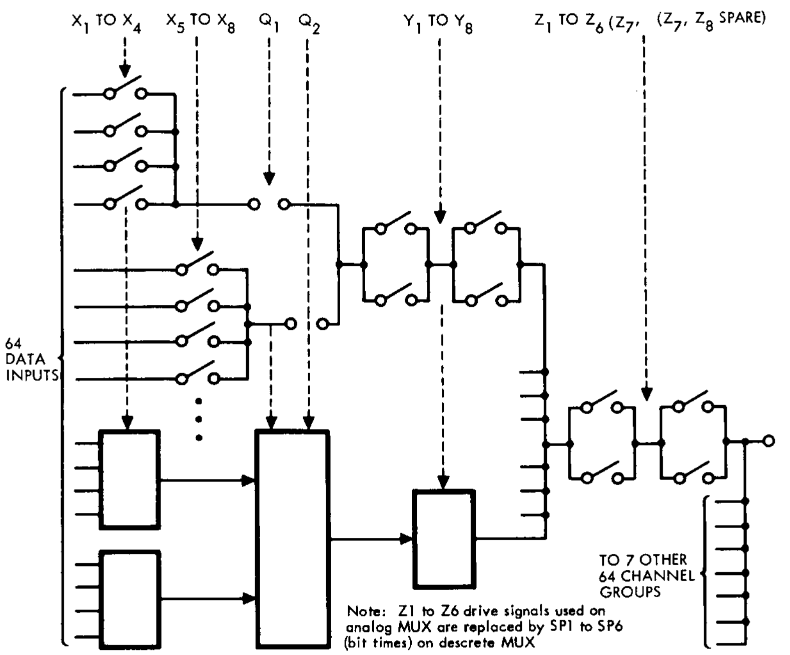


Figure 1-13.

of eight groups of 64 gates, which provides one gate for each data source. The next level isolates each four-input gate group from each other to prevent more than four inputs to be affected with any one input fault. There are 16 such gates for each group of 64 first-level gates. The next level of gating is the second-level gate. The second-level gate consists of four gates arranged in a series parallel combination to provide switch redundancy.

The analog output from the analog multiplexer board is fed into the analog-to-digital converter. The digitized data output from the analog-to-digital converter and the digital output from the digital multiplexer are combined into a single line output to the combiner. Figure 1-14 shows typical first-level gating with driver and input fault voltage protection circuits. Each input gate has a resistor at its input to limit the input current. An abnormally high-input voltage (about 16 volts) is clamped by the substrate zener at approximately 16 volts. Since the first-level MOS substrate is clamped to 16 volts,

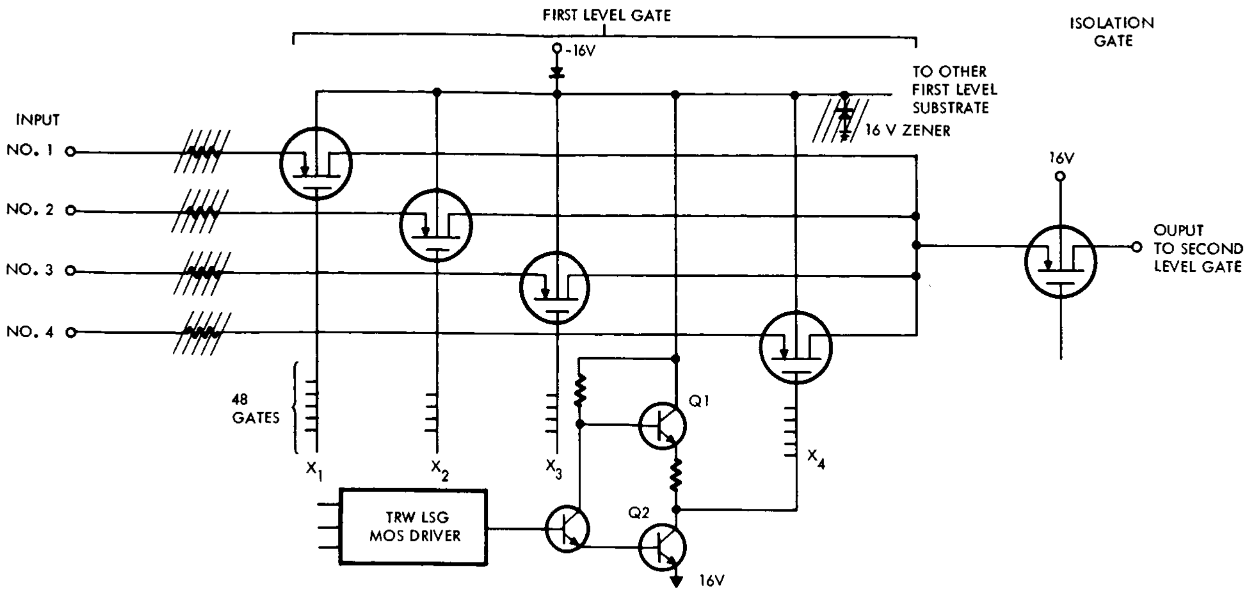


Figure 1-14

TYPICAL FIRST LEVEL GATE AND FAULT PROTECTION CIRCUIT FOR DATA MULTIPLEXER illustrates an input fault protection technique which utilizes a zener diode to clamp the first level MOS gate output voltage to prevent fault propagation through second level gates. A current limiter resistor is used at each gate input to prevent overstressing the MOS during an abnormally high input voltage condition.

the input fault cannot propagate through the second level MOS because the substrate of the second level MOS is biased by a more positive voltage. Note that if the input voltages were not clamped to a maximum of 16 volts, the isolation gate source to substrate diode would be forward biased. This would cause the other common drains of the isolation gate to be faulted and would affect seven other data inputs. In addition to the fault-voltage clamping, the fault-protection circuit also provides high-input impedance during power off conditions. For example, when power is off (assuming all voltages at ground potential), the transistor Q_1 and diodes D_1 and D_2 are back-biased, preventing a signal path to ground. This circuit provides an input impedance of greater than 1 meg when power is off. The fault protection circuit is common to analog discrete and digital multiplexers.

The MOS requires a level shift gate (LSG) driver which converts the DTL zero to +4 volts logic levels to levels suitable for driving the MOS (i.e., +16 to -16 volts). A typical LSG with its low-impedance output circuit is shown in Figure 1-14. Discrete output transistors Q_1 and Q_2 are required since the LSG does not have the required driver capability (128 MOS) to maintain the required switching speed due to charging all MOS gates and stray capacitances.

The basic MOS gate flatpack configuration for a 64-data-input group is illustrated in Figure 1-15. There are a total of six 64-input groups for the analog multiplexer (baseline system) six 64-input groups for the discrete multiplexer and one 64-input groups for the digital inputs. Each group of 64 first-level gates require 16 flatpacks, each containing four MOS gates with common outputs (drains). The next level of gates are the isolation gates which require four flatpacks, each containing four isolated MOS gates (separate drains). The next level of gating is the second-level gate which require eight flatpacks, each flatpack containing four gates. These four gates have common drains. The gate inputs from the four gates are tied together and are driven by one control signal from the programmer. The MOS sources are tied together in such a way as to form a quad redundant gate, as shown in Figure 1-13. The next level of gating is the third-level gate. One flatpack is required to perform the third-level gating for each 64-input group. The third-level gate configuration is identical to the second-level gate (redundant quad gate).

To summarize, the analog multiplexer requires 96 MOS flatpacks to perform first-level gating; 24 MOS flatpacks to perform isolation gating; 48 MOS flatpacks to perform second-level gating; and six MOS flatpacks to perform third-level gating; for a total of 174 MOS flatpacks. The discrete multiplexer also requires 174 MOS flatpacks and uses the identical configuration as the analog multiplexer. The digital multiplexer requires 32 MOS flatpacks to perform first-level gating; eight MOS flatpacks to perform isolation gating; 16 flatpacks to perform second-level gating; and two flatpacks to perform third-level gating for a total of 58 MOS flatpacks. This total includes data and enable gates. The total for all three multiplexers is 406 MOS flatpacks.

In addition to the MOS flatpacks, each multiplexer requires one current limiting resistor flatpack for each first-level MOS flatpack. Therefore, a total of 224 resistor flatpacks are required. The electrical interfaces of a 64-word multiplexer system are shown in Figure 1-16.

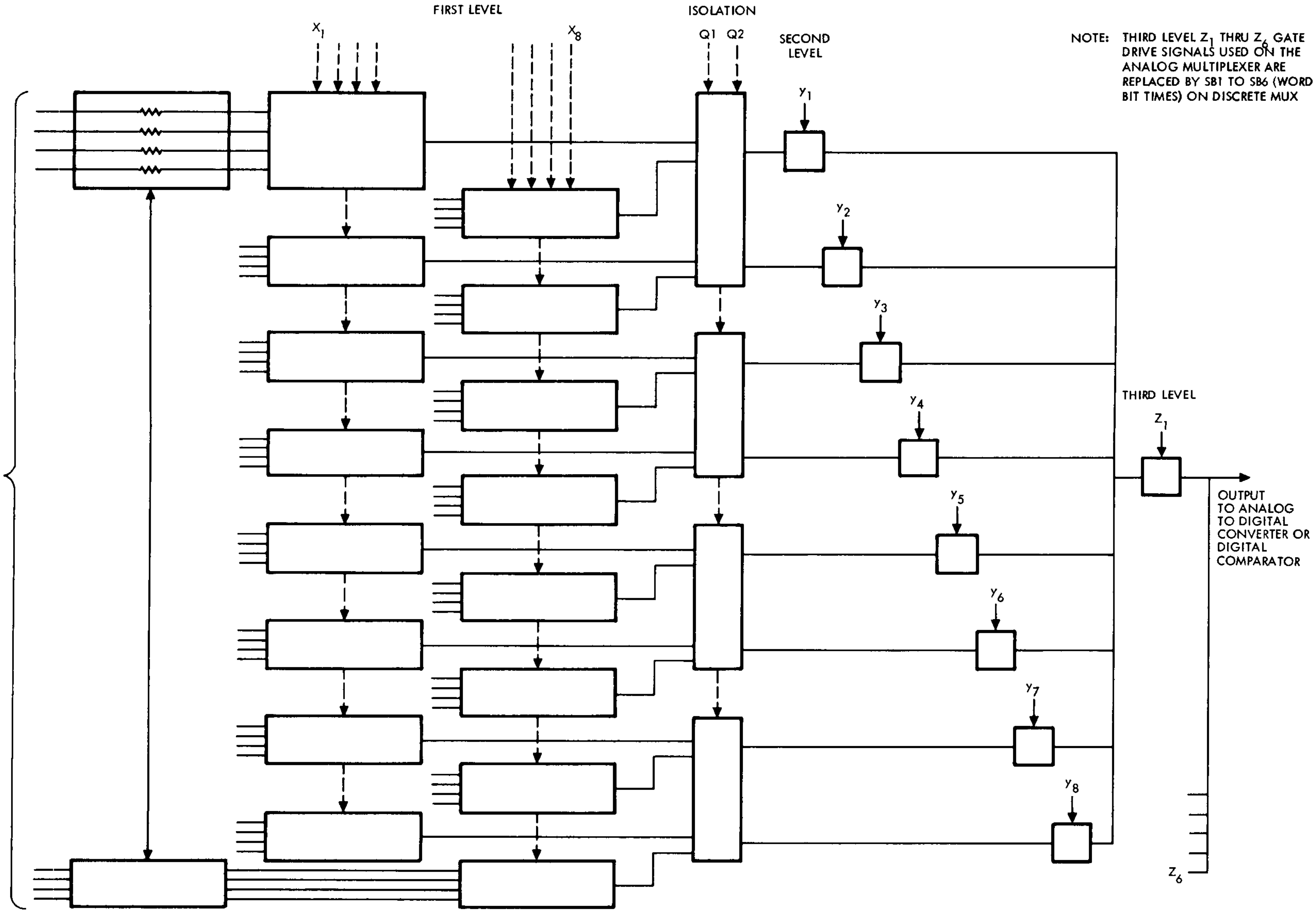


Figure 1-15
TYPICAL 64 INPUT MOS FLATPACK CONFIGURATION is shown illustrating the number of flatpacks required for each level of gating.

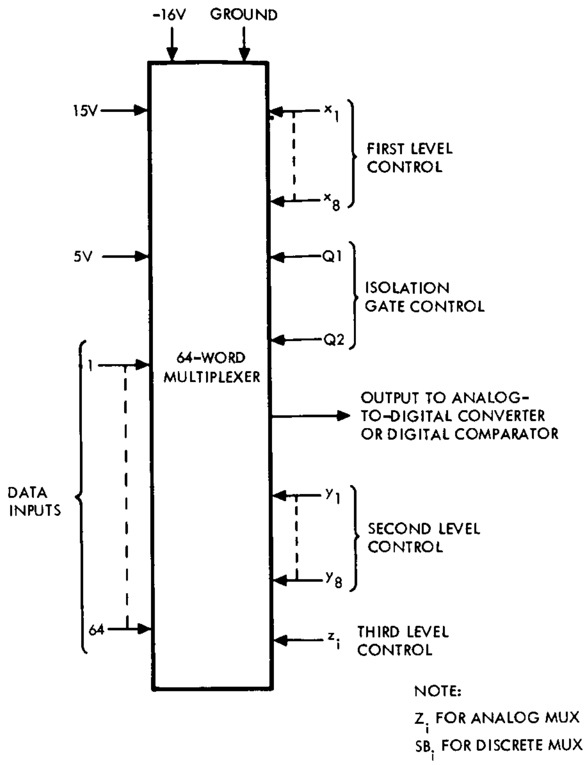


Figure 1-16
ELECTRICAL INTERFACE BLOCK DIAGRAM OF A 64 WORD MULTIPLEXER SYSTEM is shown illustrating the 64 data input and single line output. The power and required control signals are also shown for each level of the four levels of gating.

Analog Multiplexer. The engineering analog multiplexer, shown in Figure 1-17, is grouped in eight blocks of 64 inputs each. Each of the eight groups contains first-level gates, isolation gates, second-level gates, and fault-protection resistors.

Each control signal is connected in parallel to each of the six blocks of 64 inputs. For example, the X_1 control signal drives eight first-level gates in each one of the eight 64-input groups. Since there are eight groups, the X_1 control signal drives 64 gates selecting 64 analog signals. The Y_1 control signal activates one MOS gate in each of the eight blocks selecting one of the eight first-level output data signals. Therefore, the Y_1 control signal drives eight second-level gates. The isolation gates located between the first-and-second-level gates provide isolation between each first-level flatpack (four MOS gates) to prevent fault from propagating to more than four channels or inputs. The Z_1 control signals select one output from the six blocks to be fed into the analog-to-digital converter. Thus, control signals $Z_1 . Q_1 . Y_1 . Z_1$

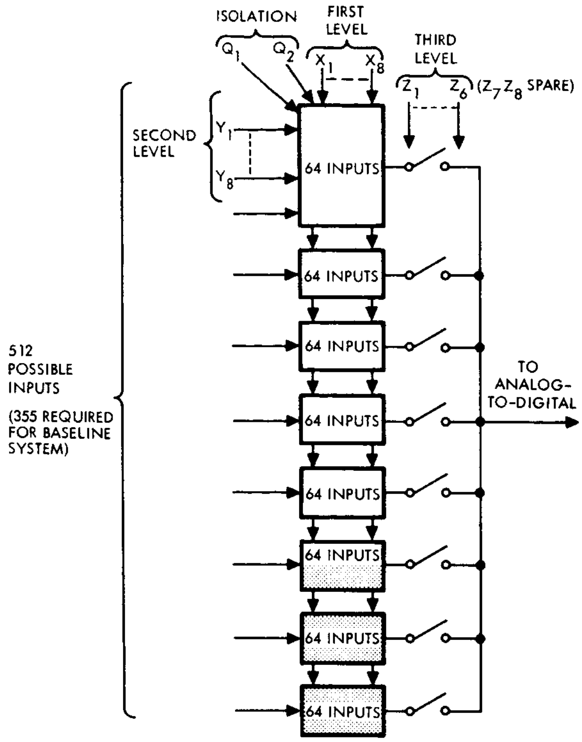


Figure 1-17
ENGINEERING ANALOG DATA MULTIPLEXER consists of six groups of 64 inputs. The additional two groups of 64 shown are not required for the baseline spacecraft.

uniquely select one of the 384 analog inputs to be fed into the analog-to-digital converter. The words contained within each group of 64 are addressable by the FFG in a sequence determined by the FFG program. When the hard wired programmer is used, inputs are selected in a predetermined sequence in a fixed format.

Discrete Multiplexer. The discrete multiplexer is shown in Figure 1-18. These gates are grouped identically to the analog multiplexer. That is, the discrete inputs are grouped in six groups of 64 inputs each.

The six groups of 64 inputs accommodate 1 bit of the 6-bit parallel word. Notice that the third-level gate is controlled by SB_1 through SB_6 which designate the 6 bits of the discrete word. The terms SB_1 correspond to the subcarrier MUX word bit times.

As in the case of the analog multiplexer, all 64 inputs of each of the six groups are addressable by the FFG in programmable sequence

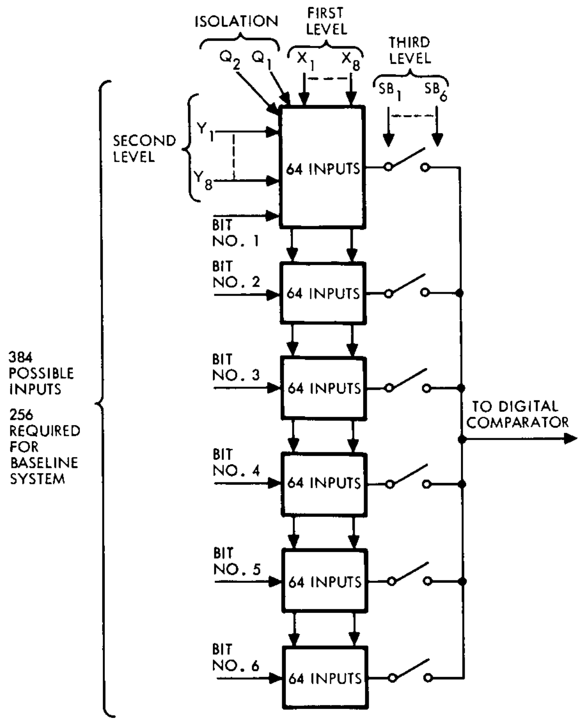


Figure 1-18
ENGINEERING DISCRETE DATA MULTIPLEXER consists of six groups of first level gates. Note, the top group of gates switch bit No. 1 of each six bit word; the second group switch bit No. 2, etc. Only 43 of the possible 64 inputs from each first level group are required for the baseline system (258 inputs).

whereas only one fixed sequence is possible with the hard wired programmer.

Digital Multiplexer. The digital multiplexer is shown in Figure 1-19. The organization is identical to the analog and discrete multiplexer except there is only one group of 64 inputs required and a corresponding one group of 64 enable outputs. The number of inputs and enable outputs can be increased by simply adding groups of 64 to the two shown in the figure.

Figure 1-20 illustrates the three multiplexers above in block diagram form together with the required addressing logic, analog-to-digital converter, digital combiner and buffer.

1.3.1.2 Flexible Format Generator

The flexible format generator consists of a 1024, 11-bit word programmable destructive readout core memory and associated logic

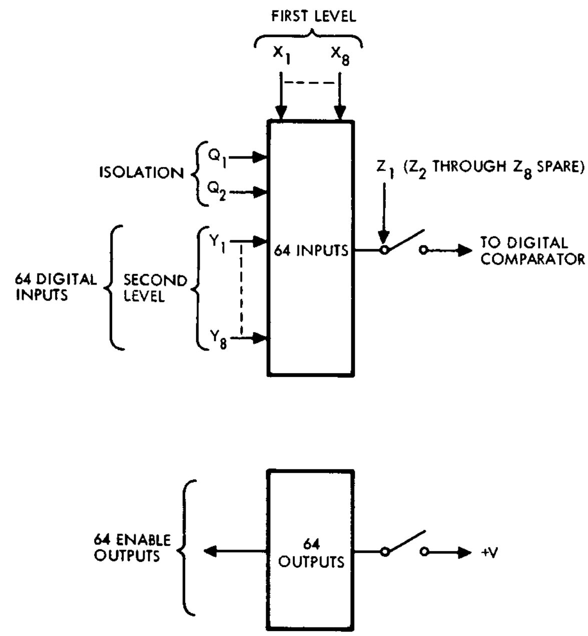


Figure 1-19
ENGINEERING DIGITAL DATA MULTIPLEXER consists of one group of 64 inputs and a group of 64 enables.

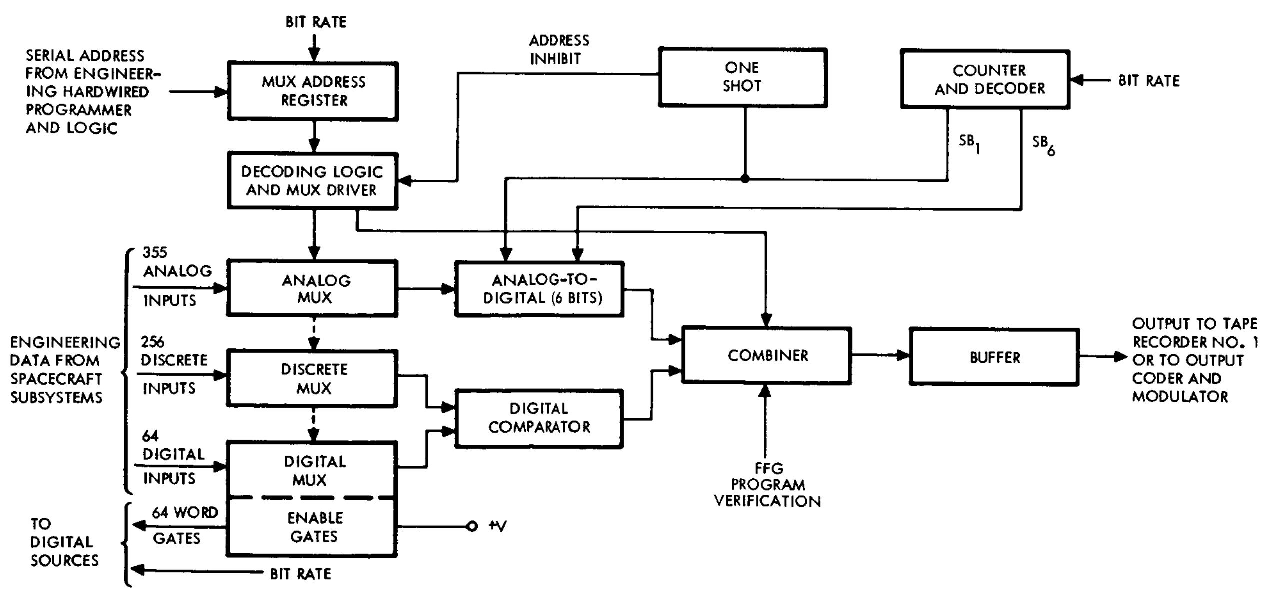


Figure 1-20
ENGINEERING DATA MULTIPLEXER illustrating the multiplexing, processing (analog-to-digital conversion) and combining of 375 engineering data inputs into a single output.

and timing circuits. It provides data selection addresses to the engineering multiplexer at approximately 85 addresses per second and provides data selection addresses to the science multiplexer at 1600 addresses per second. Data selection addresses are sequentially read out of two banks in the flexible format generator memory. One bank contains the engineering multiplexer address and the other bank contains the science multiplexer addresses. One instruction (memory location) is required for main frame words and two instructions are required for subframe words. The number of 11-bit words required to generate a given format is defined below.

$$\text{Number of words} = MFW + 3 + SFW + 2(NSF)$$

where: MFW = number of main frame words

SFW = number of subframe words

NSF = number of subframes

3 = Frame ID, frame start address, and format start address.

If two or more flexible formats contain the same subframes, the same subframe word cells can be used in these formats increasing memory utilization efficiency.

The memory is organized for coincident current, random access to permit maximum flexibility of memory operation. Each memory word is readout 11 bits per read cycle.

The average power required for the format generator at a 12.8 kb/sec bit rate is approximately 1.2 watts. Its simplified block diagram is shown in Figure 1-21. The flexible format generator basically consists of a serial input register that receives programming addresses from the command link; two primary address source registers, one for the science multiplexer addresses and one for the engineering multiplexer addresses; a secondary address source register which readdresses the memory to a jump address based upon the appropriate instruction; a core memory which consists of 1024 11-bit word cells and an input/output register which accepts programming data, serves as a temporary memory for the destructive readout/restore cycle and

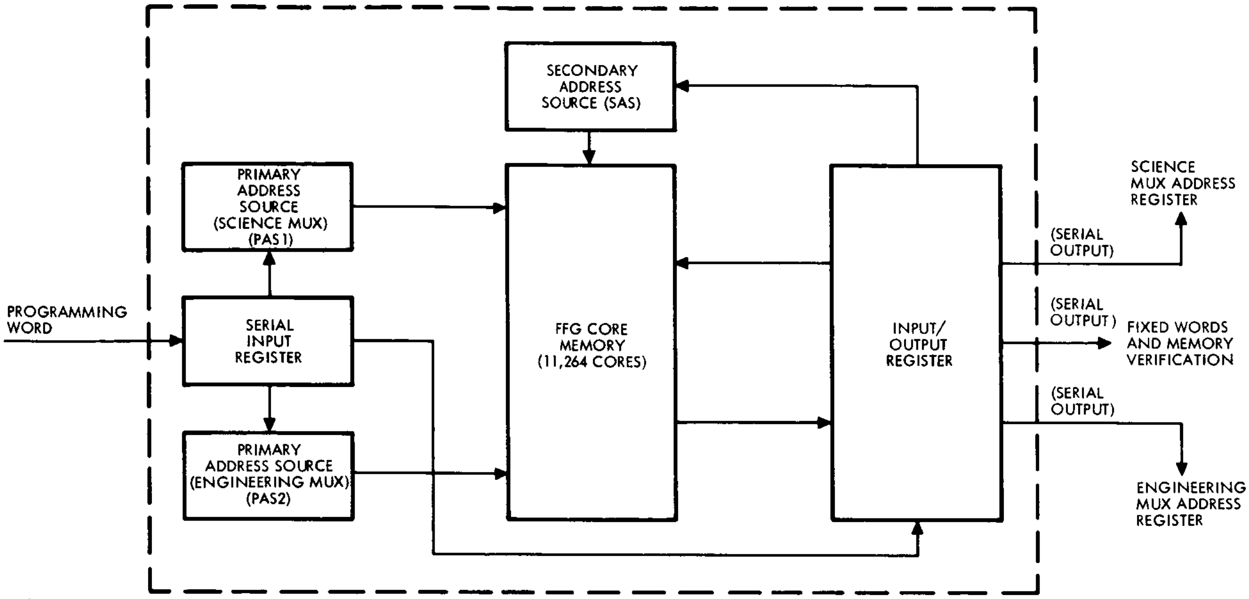


Figure 1-21

FFG BLOCK DIAGRAM illustrates the basic FFG building blocks which consist of a core stack and associated registers. Note, two primary address source registers are used PAS1 and PAS2. PAS1 selects FFG memory locations containing science multiplexer address and PAS2 selects FFG memory locations containing engineering multiplexer addresses.

serially shifts addresses to the science and engineering multiplexer address register. Address selection from the core memory is timed such that no conflict exist in the timing of memory readout cycles for the science and engineering multiplexers. Addresses that are selected from the flexible format generator core memory are serially shifted at a high rate (51.2 kHz) to either the science or engineering multiplexer address registers. The addresses are stored within each multiplexer register for their respective word times for selection of multiplexer data inputs. An inhibit signal prevents the multiplexer address registers from selecting random data inputs as the addresses are serially shifted in from the flexible format generator.

The contents of the flexible format generator are readout, formated and transmitted upon ground command for memory content verification. The hardwired programmer performs the required formating in this mode.

The flexible format generator generates a frame rate and subframe rate signal used for timing operations within the telemetry and data storage subsystem and also supplied to the experiment packages for timing experiment operations. Frame synchronization words (Barker codes) are read out of the flexible format generator and formated with the transmission data stream at the beginning of each frame.

The flexible format generator core memory is fabricated using standard core memory techniques. The FFG power supply is designed to store enough power to complete a read restore cycle should power be interrupted during the cycle. (The use and detailed implementation of a telemetry system using the FFG concept is presented in a NASA proposal RFTP No. BG731-49-7-325TP (stored program data processor) dated 12 June 1967.

1.3.1.3 Hardwired Programmers and Logic

There is one hardwired programmer and logic element associated with the engineering data multiplexer and one with the PSP remote multiplexer (science). Both hardwired programmers consist of two counters containing six flip-flops. The first counter counts binarially through 64 main frame words and second counter binarially through 64 subframes. These counters are used to supply addresses to the multiplexer address registers, via the multiplexer control logic, to select data inputs in a fixed format. The purpose of the hardwired programmers is to provide limited formatting capability as a backup to the FFG and to generate a format for the verification of the FFG memory. Figure 1-22 illustrates the elements of the engineering hardwired programmer and logic. The diagram illustrates the basic logic required to serialize addresses from either the FFG or the hardwired programmer. The addressing of the multiplexer via the flexible format generator is straight forward since there is a unique address for every multiplexer input and there are no special considerations necessary to handle subframe words. The hardwired programmer, on the other hand, needs special consideration in the area of subframe words necessitating the use of a special words decoder. The special word decoder identifies which of the preselected words in the frame are subframe words and also determines whether the addressed word is an analog, discrete or digital word. When a subframe word is detected, the special word decoder logically substitutes the output states of the subframe counter for the output states of the mainframe

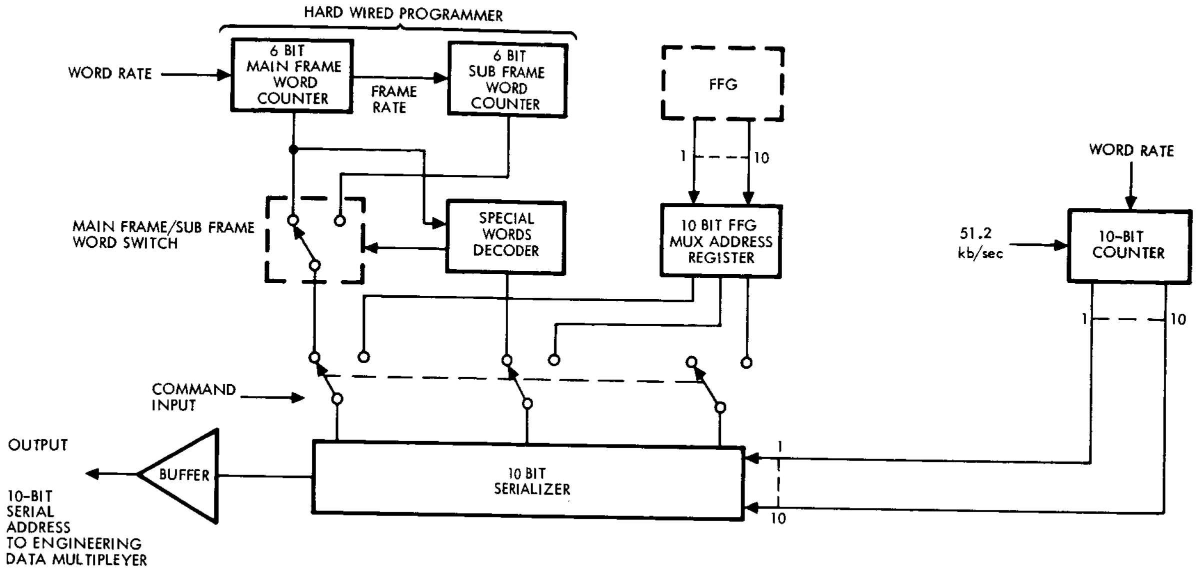


Figure 1-22
ENGINEERING HARDWIRED PROGRAMMER AND LOGIC illustrates the basic logic switching required to decode addresses from the FFG and the hardwired programmer for driving the analog, discrete and digital data multiplexers.

counter. This technique allows the time sharing of MOS drivers and greatly simplifies MOS organization.

Figure 1-23 illustrates the elements of the science hardwired programmer and logic which operates identically to the programmer discussed above.

1.3.1.4 Analog to Digital Converter

Two analog-to-digital quantizations are required for the telemetry and data storage subsystem: 6 bits for engineering and video data, and 8 bits for science data. The analog-to-digital converters use the efficient, proven successive approximation technique. Each converter consists of a comparator followed by logic and decoding which creates a trial output voltage that in turn is fed back to the comparator for comparison with the input analog voltage. This process is performed six or eight times, depending upon whether a 6- or 8-bit digitization is required. In the successive approximation technique used, each 6- or 8-bit conversion is done at the last few microseconds of each bit time.

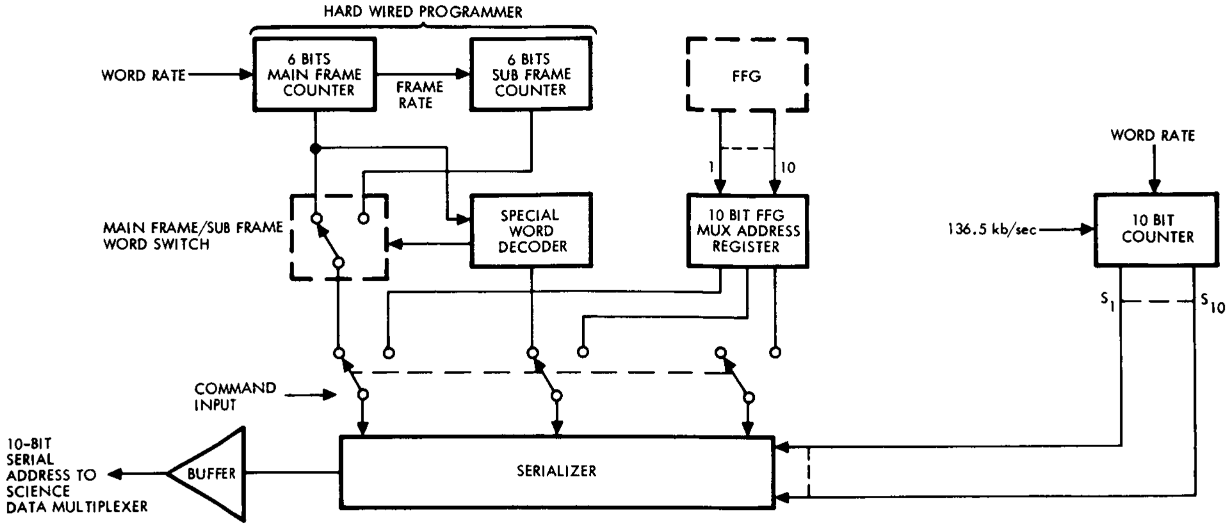


Figure 1-23
SCIENCE HARDWIRED PROGRAMMER AND LOGIC performs the required switching between the FFG mux address register and the science hardwired programmer; serializes the address bits and shifts these address bits to the PSP remote data multiplexer via one line at the 136.5 kb/sec bit rate.

The block diagram of the converter is shown in Figure 1-24. The ladder-adder can produce analog output voltages up to 5.1 volts, in steps of 20 mv, in response to binary input numbers defined by the current sources. The ladder-adder output voltage is compared against the selected analog input voltage at successive bit times, B_1 through B_8, by the comparator. The results of these comparisons are sensed by the control logic, which operates on flip-flops M_1 through M_8. For example, during the first bit time, the analog is compared against a reference voltage equal to 1/2 of full scale. The comparator output is true or false, depending upon whether the reference voltage is higher or lower than the analog voltage, and it will set the next reference voltage to either 1/2 + 1/4 or 1/2 - 1/4 of full scale at the second bit time. The latter-adder input currents are progressively changed in this way to bring the output voltage closer and closer to the selected analog voltage. When the ladder-adder output voltage is within 20 millivolts of the selected input voltage, the flip-flop states define the binary equivalent at the end of the conversion time. The output of the 8-bit register is read out serially by bit times B_1 through B_8.

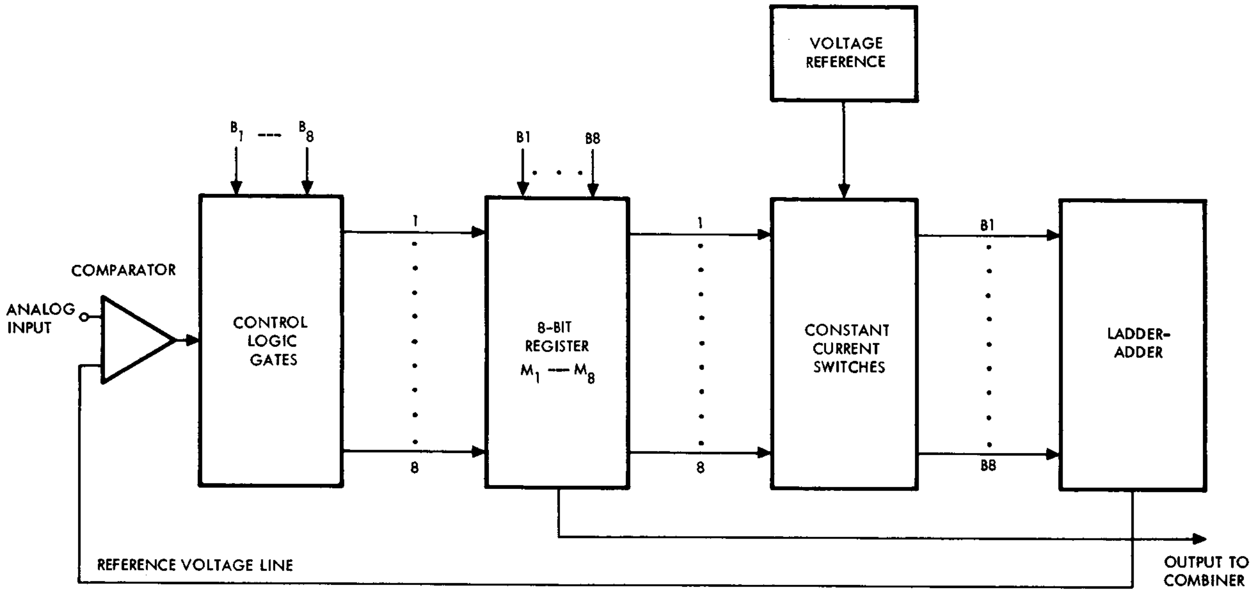


Figure 1-24
ANALOG-TO-DIGITAL CONVERTER block diagram illustrates the circuitry required for the successive approximation analog-to-digital system. Eight constant current generators are switched on or off successively during the 8-bit times B1 through B8 to generate a voltage within the ladder-adder (resistor network) which approximates the input voltage. The state of each current source is stored within the 8-bit register M1 through M8 and read out as the digitized word.

Power gating shuts off logic and current sources to conserve power. Power is turned on only at the last few microseconds of each bit time prior to each quantization. The unity gain amplifier is adjusted to an offset voltage equivalent to one quantization level (20 mv for eight bit system). Thus, for a zero volt input, the digitized code is 00000001, and for less than zero, the code is all zeros. The full-scale voltage is defined as 254 x 20 mv = 5.08 volts, which is the threshold point between the 11111110 and 11111111 output codes. The offset at the unity gain amplifier is to obtain a positive quantization output for zero volts input (to distinguish from all zero indication of a nonoperating analog to digital).

Monolithic integrated circuits will be used for Voyager converters to increase reliability and reduce size and weight.

1.3.1.5 Biorthogonal Coder

Coding of the telemetry data offers a way of increasing the information rate at the expense of transmission bandwidth without unduly complicating the spacecraft telemetry subsystem. Coding requires a substantial bandwidth expansion, however, this bandwidth capability is already available for the Voyager application due to the pseudo random-noise ranging signal requirements. By time sharing the high rate transmission channel between the mapping data and the PRN ranging signal. A two-sided transmission bandwidth of approximately 2 MHz for the mapping data is available. This is more than adequate for coded transmission at the highest data rate, 51.2 kb/sec.

Two basic coding approaches have been considered and are described in some detail in Appendix A. Sequential or bit by bit coding has the advantage that it requires a bandwidth expansion of only two to one with theoretical information rate improvements of approximately 4 db (2-1/2 times) compared to equivalent coherent PSK systems. The disadvantage of this technique is that the statistical nature of the decoding process dictates a variable decoding rate which may not be compatible with on-line (real-time) data processing at the Voyager transmission rates. The other disadvantage is that it is a rather recent development and does not have the proven performance characteristics of the block coding systems. A sequential coding system is in the development phase for use at lower data rates on upcoming Pioneer launches. Experience gained from this and other development programs may make the sequential coding approach more attractive as improved decoding techniques and additional data on performance characteristics are obtained.

Biorthogonal block coding using the selected 32, 6 biorthogonal code offers a theoretical improvement approximately equivalent to the sequential code (4 db). It has the disadvantage that it requires approximately five times the transmission bandwidth of coherent PSK. There are two primary advantages to the biorthogonal code. The first is that it operates at a fixed decoding rate and is compatible with real time decoding at the Voyager transmission rates. It also has the advantage that a substantial amount of experience in these systems has been gained in the last several years including a system (Digilock) which was actually developed and flight tested as early as 1960.

Because of these advantages, the biorthogonal system has been chosen as the most conservative approach. The bandwidth expansion does not represent a constraint for the basic Voyager application since sufficient transmission bandwidth is available. For an upgraded system employing higher data rates, this may not be true and for that application further consideration of the sequential approach will be taken.

The two telemetry links (main carrier and subcarrier) each includes a biorthogonal coder that encodes each six data bits into a highly redundant 32-bit word for transmission. As the code is comma free, word sync is automatically obtained from the bit stream.

The basic code is generated in accordance with the equation below.

$$(a_0, a_1, a_2, a_3, a_4, a_5) \rightarrow a_0 \cdot \bar{u}_0 + a_1 \cdot \bar{u}_1 + \ldots a_5 \bar{u}_5$$

The a_0 through a_5 terms represent six data bits which are coded into a 32-bit word when operated on by basis vectors u_0 through u_5. The basis vectors are defined below.

$$\overbrace{\begin{array}{l} \bar{u}_0 - 0\,1 \ldots\ldots\ldots\ldots\ldots\ldots\ldots 0\,1 \\ \bar{u}_1 - 0\,0\,1\,1 \ldots\ldots\ldots\ldots\ldots\ldots 0\,0\,1\,1 \\ \bar{u}_2 - 0\,0\,0\,0\,1\,1\,1\,1 \ldots\ldots\ldots\ldots\ldots 0\,0\,0\,0\,1\,1\,1\,1 \\ \bar{u}_3 - 0\,0\,0\,0\,0\,0\,0\,0\,1\,1\,1\,1\,1\,1\,1\,1 \ldots 0\,0\,0\,0\,0\,0\,0\,0\,1\,1\,1\,1\,1\,1\,1\,1 \end{array}}^{32}$$

$$\bar{u}_4 - \overbrace{0\,0\,0\,0 \ldots\ldots\ldots 0\,0\,0\,0}^{16} \quad \overbrace{1\,1\,1\,1 \ldots\ldots\ldots 1\,1\,1\,1}^{16}$$

$$\overbrace{\bar{u}_5 - 1\,1\,1\,1\,1\,1\,1\,1 \ldots\ldots\ldots\ldots\ldots 1\,1\,1\,1\,1\,1\,1\,1}^{32}$$

Before transmission, the following comma free vector is added to each 32-bit word obtained by the above operations.

1000 1101 1101 0100 0010 0101 1001 1111

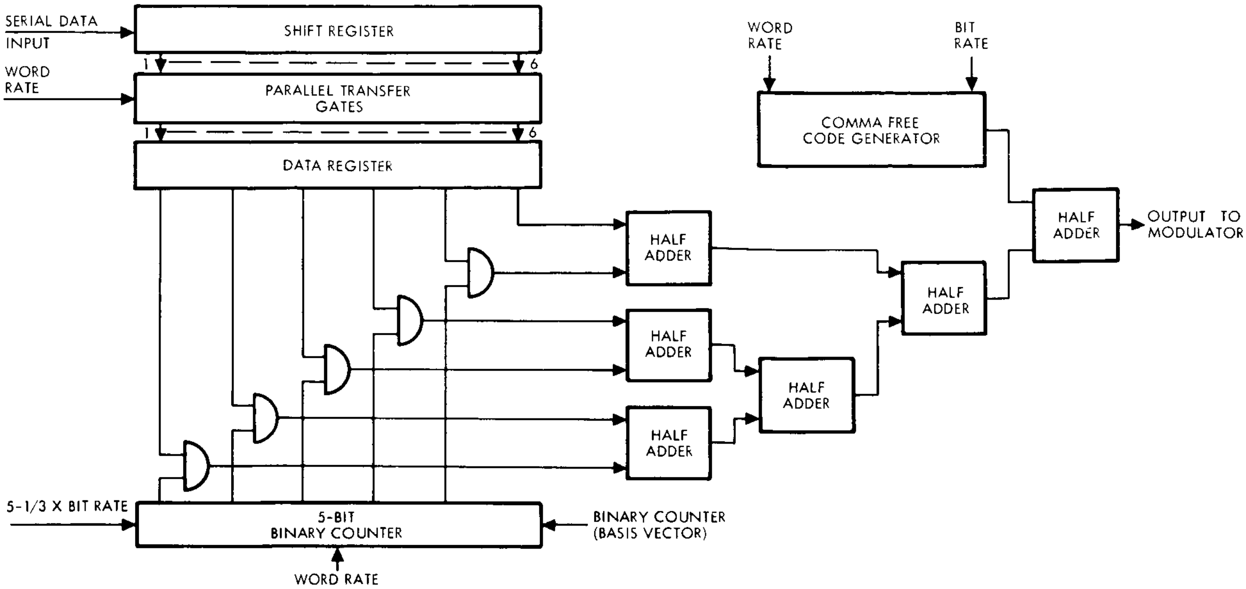


Figure 1-25
BIORTHOGONAL CODER BLOCK DIAGRAM illustrates the four registers and half adders which constitute the required coding logic. Thirty-two coded bits result for each 6-bit data word.

The implementation of the biorthogonal code is illustrated by Figure 1-25. NRZ data enters the shift register in a continuous stream and is parallel shifted into the data register at the end of the sixth bit. The corresponding outputs from the data register and the basis vector register are added together then all summed together by half adders as shown.

The last adder outputs the biorthogonal coded data at 5-1/3 times the prevailing data bit rate. For example, an uncoded data rate of 512 bits/sec becomes 2730-2/3 bits/sec when coded for transmission.

1.3.1.6 Subcarrier Oscillator

Two crystal controlled oscillators will be used to generate the clocks used for the biphase modulators. One oscillator will generate 1.024 MHz for normal subcarrier modulation at the output data rate of 512 bits/sec and the other will be 2.048 kHz for emergency subcarrier operation at 8 bits/sec.

1.3.1.7 NRZ Data Modulators

Formatted data for the subcarrier is biphase modulated before it passes to the transmitter. The biphase modulator logic and typical waveforms are shown in Figure 1-26. A low-pass filter limits the high-frequency components to 1.024 MHz. Formatted data for the main carrier channel and video data are split-phase modulated using the system shown in Figure 1-27. The split-phase modulation is performed by a half-addition of the nonreturn-to-zero data and a square wave at the bit rate.

1.3.1.8 Timing Generator

The timing generator is illustrated by Figure 1-28. A master clock frequency of 819200 Hz is required so that the highest uncoded data bit rate (51200 Hz) and coded data bit rate (273066-2/3 Hz) can be obtained in the exact synchronization required by the biorthogonal coder. All other coded and uncoded rates are then obtained by direct countdown as shown. The particular rates for a given mode are selected on command from the computer and sequencer.

1.3.1.9 Video Tape Recorder Switching

A network switches video inputs between the four tape units by the arrangement shown in Figure 1-29. Tape unit No. 6 can replace

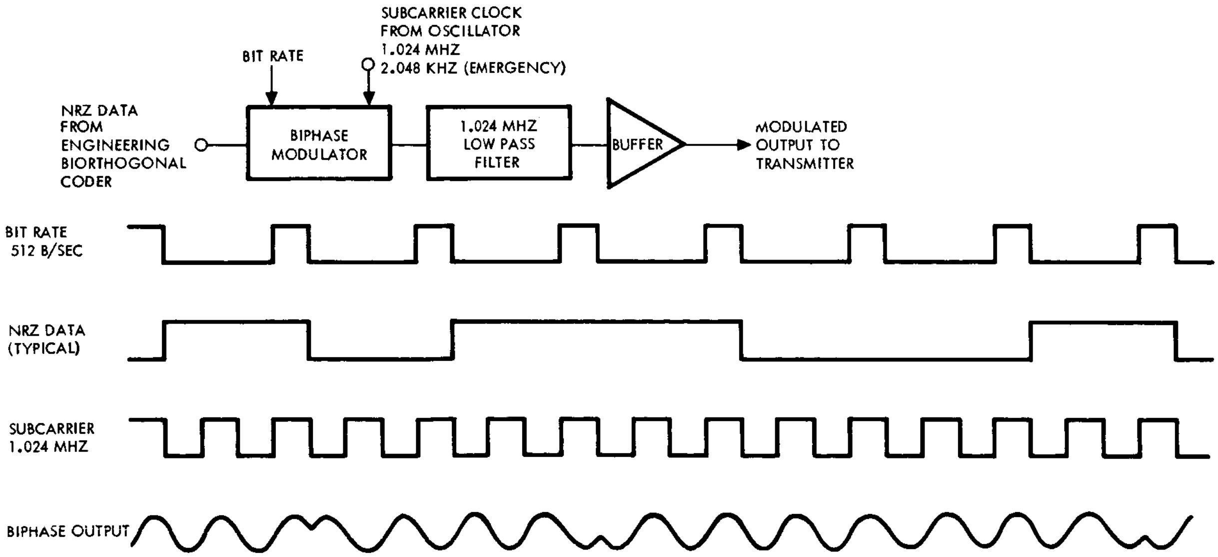


Figure 1-26
BIPHASE MODULATOR AND TIMING illustrates NRZ-C (change) type modulation for 1.024 MHz and 2.048 KHz (emergency) subcarriers and the required circuitry. The subcarrier rate to bit rate is shown as a 2:1 ratio for clarity, however, in reality the ratio is 2000:1.

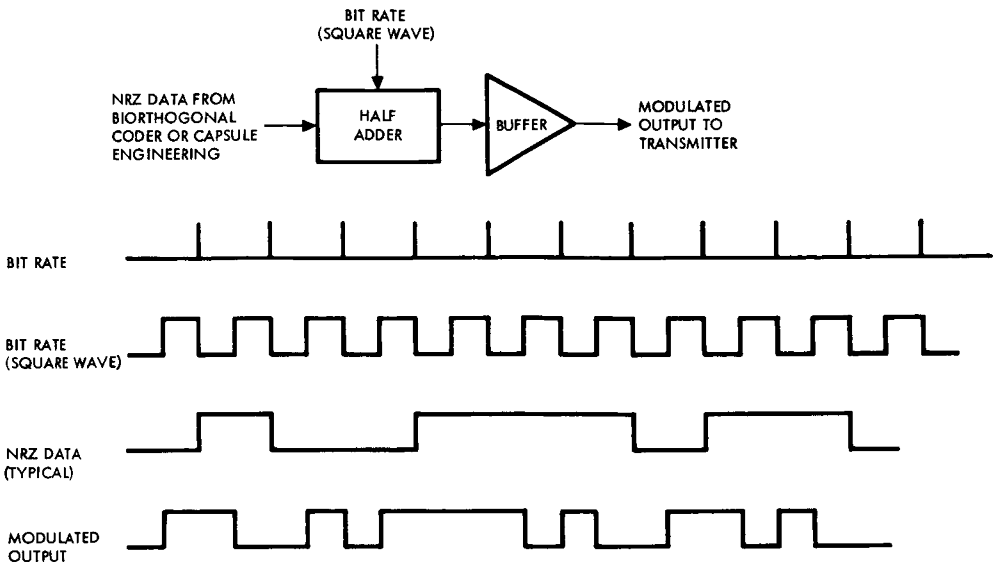


Figure 1-27
SPLITPHASE MODULATION is used to modulate the main carrier input to the transmitter to insure at least one transition each two bit times.

either tape unit No. 3, No. 4 or No. 5, all of which are used simultaneously during mapping operations. Redundancy is provided for tape unit No. 6 by switching the high resolution, low resolution or capsule TV data to No. 5 should No. 6 fail.

1.3.2 PSP Remote Data Multiplexer (on Planetary Scan Platform)

This remote multiplexer for science data is similar in organization to the engineering multiplexer except that it is much smaller and requires the selection of only 64 analog and 16 digital inputs. An identical remote multiplexer will be used on the upgraded spacecraft to select data from spacecraft body-mounted experiments.

Figure 1-30 shows the multiplexer organization and circuitry. The 8-bit serial address from the telemetry data handling unit is serially shifted to the remote data multiplexer at a 51.2 kb/sec rate. Note, only eight of the 11 FFG address bits are required for the remote multiplexer. An address-inhibit signal prevents random addressing as the address shifts into the address register. An 8-bit counter provides synchronized bit time pulses to the analog-to-digital converter for timing of the conversion process. All necessary address decoding is performed by 18 level shift gates within the decoding logic

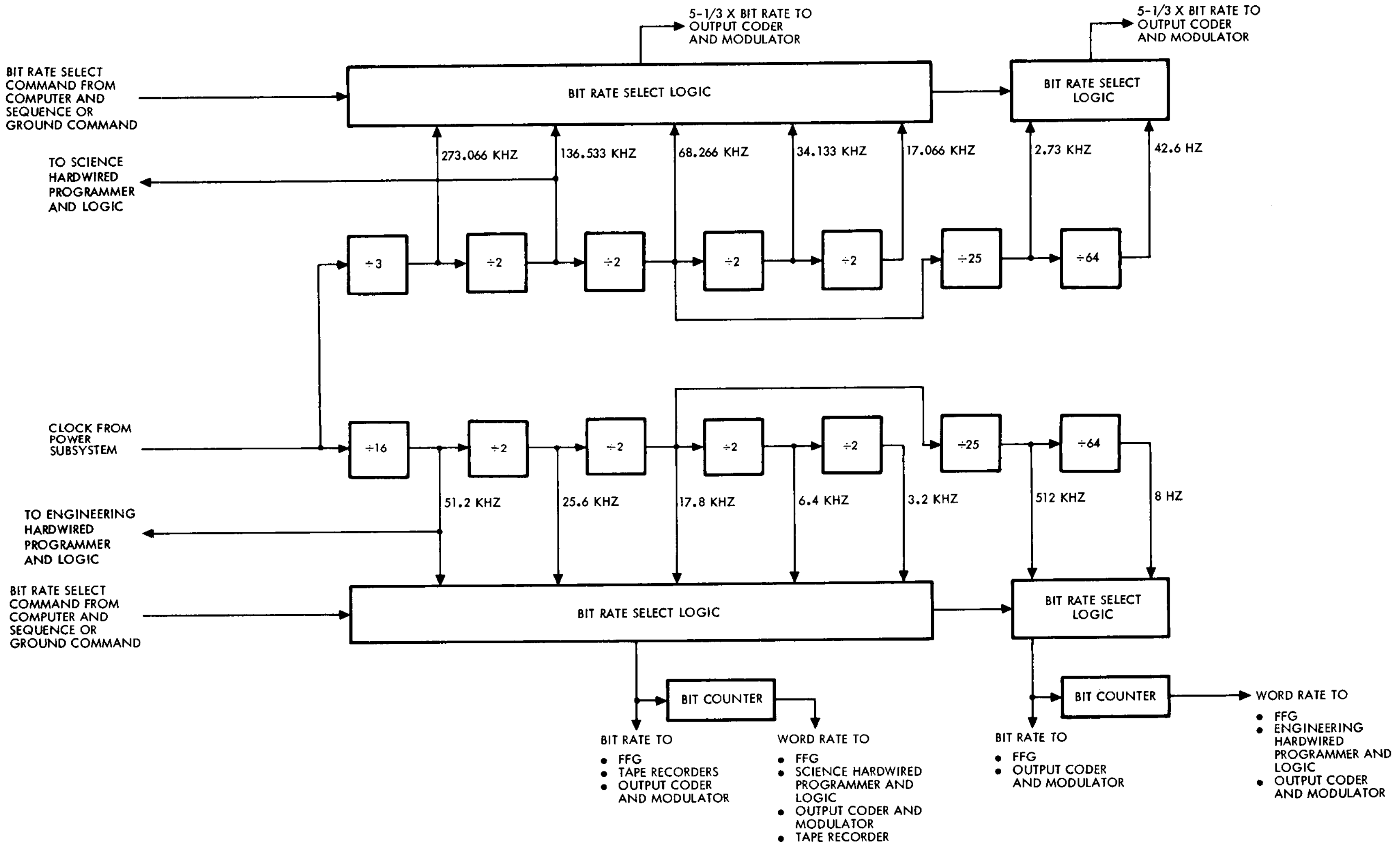


Figure 1-28
TIMING GENERATOR generates the various main and subcarrier clocks and word rate pulses for the FFG, tape recorders, output coder and modulator and engineering and science hardwired programmer and logic.

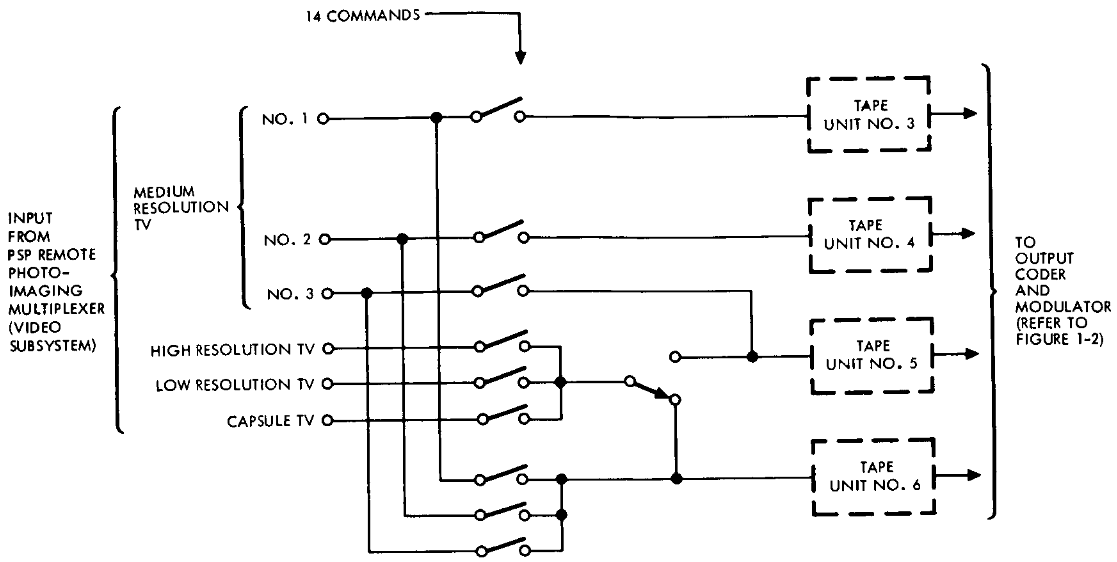


Figure 1-29
VIDEO TAPE UNIT INPUT SWITCHING is used to provide a means to switch to another recorder should a recorder fail.

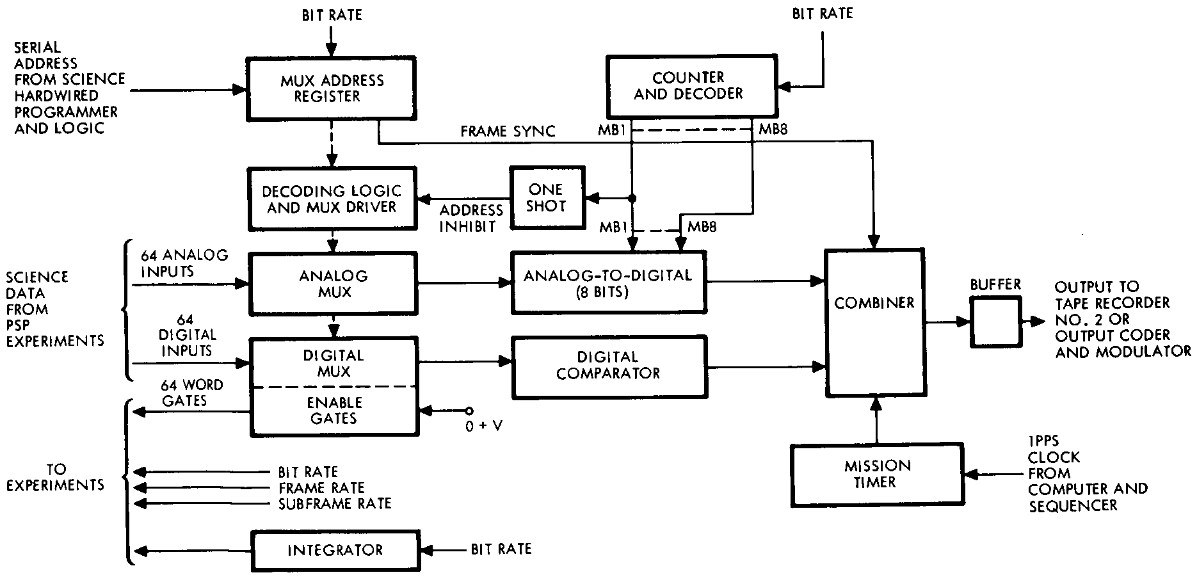


Figure 1-30
PSP REMOTE DATA MULTIPLEXER is located within the PSP and accepts serial address information from the science hardwired programmer and logic, and decodes these addresses to select data inputs according to the FFG stored program or the science hardwired programmer. The selected analog inputs are encoded to eight bit accuracy and time multiplexed with sampled digital data, buffered and sent to the telemetry data handling unit via a single line.

that drive the metal oxide semiconductor gates. The output from the analog-to-digital converter and the digital comparator are gated to the buffer driving the data output line going to the central telemetry data handling unit.

1.3.3 Remote Photo-Imaging Multiplexer

The remote photo-imaging multiplexers are located within the PSP and for the hypothetical camera system consist of five video data encoders of the type shown in Figure 1-31, one for each television camera. Individual video data encoders are required since several of the television cameras must operate simultaneously.

One video signal and typically 22 engineering data are multiplexed together with frame synchronization codes to produce a photo-imaging picture format. The frame synchronization generator produces a 24-bit pseudorandom code at the beginning of each horizontal-scan line and enters the engineering measurements between each picture. As shown in the figure, a 6-bit counter generates bit time pulses at a 697 kb/sec rate for the analog-to-digital converter. A programmer generates outputs which are decoded by the level shift gate, which in turn drive the metal oxide semiconductor gates. The programmer also generates four word times for the frame sync generator. The digitized video and engineering data and frame sync data is combined and buffered for transmission to the central telemetry data handling unit via one line per television camera for storage in the video tape recorders. Engineering measurements from each television camera will be repeated until the next picture-taking sequence. The remote photo-imaging multiplexer timing is independent of the telemetry data handling unit; the only interface to the telemetry system is to the tape recorders.

1.3.4 Data Storage Units

The magnetic tape recorders used for television and capsule data and for science and engineering data storage are essentially the same type, configured for different tape lengths and different record/playback speeds. A preliminary specification for the units is shown in Figure 1-32.

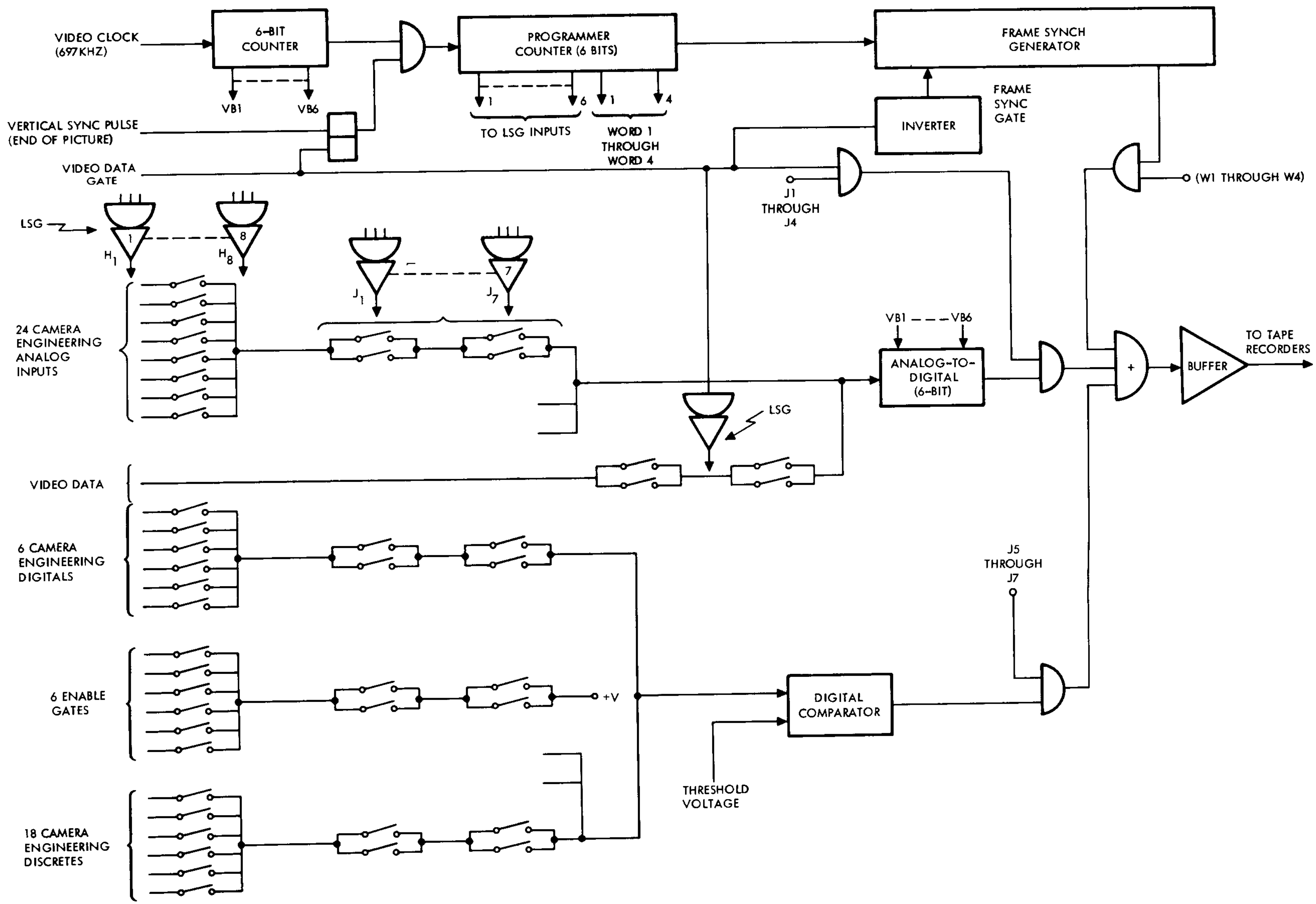


Figure 1-31
PSP REMOTE PHOTOIMAGING MULTIPLEXER-VIDEO DATA ENCODER time multiplexes telemetry frame sync at the start of each picture horizontal scan line and also time multiplexes camera engineering measurements into the video data stream at the end of each picture.

PRELIMINARY SPECIFICATION

Tape Recorder Data Flow, Motor Drive and Motor Speed Control

Performance Characteristics

	Television and Capsule	Science and Engineering
Capacity (bits)	9×10^8	7.5×10^7
Input Rate (kb/sec)	100, 697	0.512, 12.8
Output Rate (kb/sec)	51.2, 25.6 12.8, 6.4, 3.2	51.2, 25.6 12.8, 6.4, 3.2
Tape Width (in.)	0.5	0.25
Tape Length (ft)	3000	1600
Tracks	14*	4**
Bit Density (bits/in.)	2330	2000
Record Method	Biphase	Biphase
Record Speed (in./sec)	7.2, 50	0.256, 6.4
Playback Speed (in./sec)	3.7, 1.85, 0.92, 0.46, 0.23	25.6, 12.8 6.4, 3.2, 1.6
Bit Dropout Rate (bits)	1 in 10^5	1 in 10^5
Power: Record (watts)	20	12
Playback (watts)	15	15
Input Voltages	+50 VDC, 400 Hz	+50 VDC, 400 Hz

Physical Characteristics

Volume (in.^{3})	650	600
Weight (lb)	20	18

1) Playback will be synchronous with telemetry data rate
2) End-of-tape will be provided for recording and playback
3) Commands will be: record, playback, speed select
4) Electromechanical elements will be in a pressurized container
5) The angular momentum change will not exceed ±0.1 ft-lb-sec between zero and full speed
6) Telemetry monitoring points: temperature, pressure, status

* One track clock, six tracks data then tracks switched to provide this capacity again in opposite direction of tape movement (track switching).

**One track clock, one track data then tracks switched to provide this capacity again in opposite direction of tape movement (track switching).

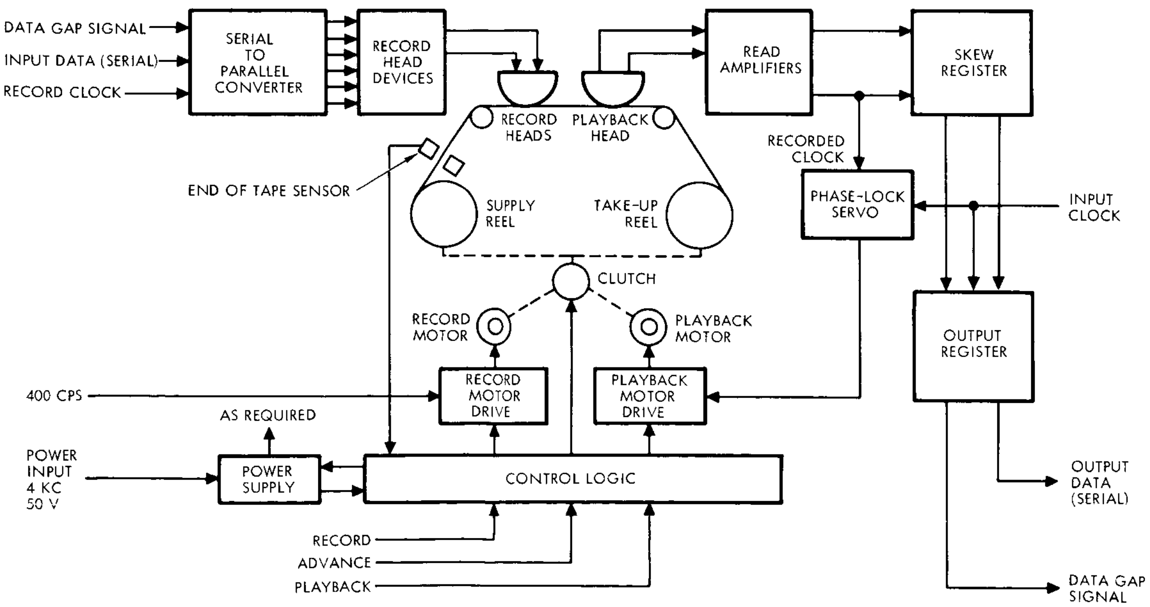


Figure 1-32

The magnetic tape recorders are of reel-to-reel configuration, capable of recording and reproducing serial digital information. They are sealed units containing a tape transport mechanism, magnetic tape and reels, record and reproduce circuits, motor drive circuits, and other electronics required to control or buffer inputs and outputs.

1.3.4.1 Tape Transport

The tape transport recommended for handling the magnetic recording tape uses the peripheral drive system (Figure 1-33) for tape drive and tensioning. Widely used in satellite recorders, this drive has the advantages of simplicity, low power, and an almost constant tape tension over the full length of magnetic tape. An in-line reel-to-reel system is used to achieve the least complex system.

Flangeless hubs store the tape. This technique has proven to be reliable and simple. It also allows the hubs of the two reels to be located closer together than if flanged reels were used, reducing overall size.

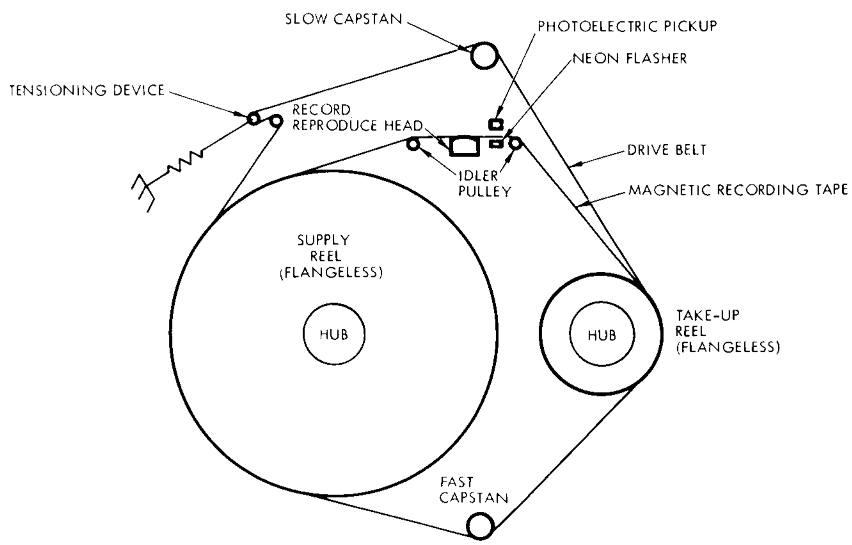


Figure 1-33
TAPE RECORDER PERIPHERAL DRIVE system is used where differential capstans drive a belt which rides on the tape for the purpose of driving the reels.

Capstans turning at different speeds drive the peripheral belt across the reels, providing a faster speed to the takeup reel. This develops the tape tension across the heads and the tape reel winding tension.

The differential capstans are driven by one of two motors, one for recording, the other for playback. The selected motor is coupled to one of two capstans through a clutch arrangement. Differential capstan speeds are obtained by a slightly different pulley dimension on the two capstans.

End of tape will be detected by a flashing neon bulb working in conjunction with a photoelectric pickup. The end portion of the tape is stripped of magnetic oxide to allow the photoelectric detector to sense the neon bulb flash when the tape end is reached. This technique has been used by several tape recorder manufacturers and has proven reliable in life testing. To prevent loss of tape in the event of failure, the tape end is fastened to the reel hub. In the event of an end-of-tape sensor failure, the unit will not be damaged, but the motor will stall-out without damage to the motor or drive electronics.

1.3.4.2 Electrical Design

Write Mode. The serial nonreturn-to-zero input data is shifted into a 6-bit flip-flop register for parallel transfer to the write amplifier. The data is transferred to the tape via the write amplifiers under the control of the word clock. This requires three inputs from the spacecraft system: a data bit signal, a word clock, and a bit clock. Recording the write clock on tape provides a synchronizing signal during the playback mode.

A phase-shifted clock can be used for writing data so that a clock which occurs in the center of a data bit is stored instead of a clock in phase synchronization with the data. This allows easy implementation of a strobe pulse for reading data out of the read amplifiers during the read mode.

Read Mode. To provide data output synchronization between tape recorder data and the spacecraft clock, a phase-lock servo loop is used in the read mode to control the motor speed, hence the tape speed. A small buffer register at the output buffer insures a constant flow of output data in the presence of small response time errors of the servo loop.

Speed Control. Using the word-one bit-one shift pulse as a toggle input to one flip-flop, and the read-in word two pulse as a toggle input to a second flip-flop, the outputs of these two flip-flops provide inputs to a phase detector to generate a DC error signal. This DC error signal represents the time positional error of the tape, which is used as an input to a voltage controlled oscillator to develop a frequency change. The oscillator output is then shaped into a square wave and counted down so as to produce two 400 cps square waves time displaced by 90 electrical degrees. These signals then, via power amplifiers, are applied to the two windings of a hysteresis motor. As phase variations which deviate from 180 electrical degrees develop between the output of the toggle flip flops, the motor speed is altered to bring the phase shift back into the correct 180 degree relationship. Figure 1-34 shows a block diagram of the technique described.

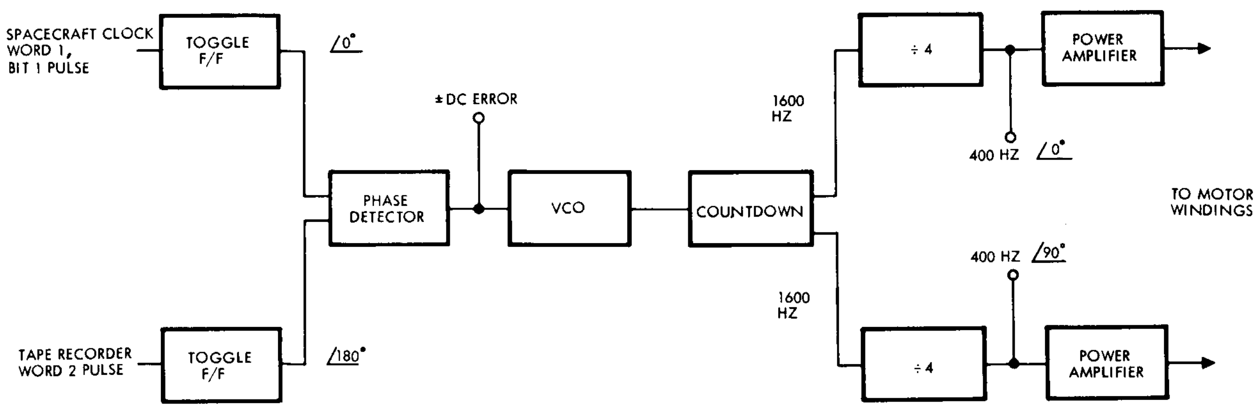


Figure 1-34
MOTOR SPEED SERVO SYSTEM uses a phase detector to detect time difference between data bits and spacecraft clock to generate a DC error signal to control a variable controlled oscillator (VCO) to develop a frequency change in the 400 Hz motor power.

At startup there is a continuous change in the error signal as the system approaches the phase lock condition. The DC error voltage is small enough (representing say 400 ± 1 cycles/sec) that the motor approaches the desired speed. As it does, the servo control loop comes out of saturation and takes over control of the motor speed.

Once the servo loop is in control, the correct time relationship exists between the spacecraft read out clock and the tape recorder read-in clock. There is, therefore, no need for special start logic in the data buffering logic.

The read electronics consist of read amplifiers monitoring the output of the read heads. The amplifiers load a 6-bit shift register with the stored data. The shift register used in the write mode is shared with the shift register used in the read mode.

Speed Change. Speed change can be accomplished by changing the motor input synchronizing frequency and the spacecraft data clock inputs during the Read mode. This would be accomplished under the control of the spacecraft command system. However, the four-speed playback requirements necessitate a development effort to find out if it is better to use a motor operating at four speeds, or a single-speed motor with a mechanical switching arrangement to obtain the four speeds.

Erase Technique. Erasing by writing over previously recorded data is recommended since it does not require any erase circuitry. To provide an adequate signal-to-noise ratio, a biphase saturation format is used.

1.3.4.3 Data Storage Development Problem Areas

The primary problems would most likely occur with the television and capsule recorder. There is a significant interrelationship between tape recorder characteristics. Fast tape speeds (50 ips or above) or long tape lengths (greater than 1800 feet) result in more head, tape, and other wear, which affects reliability and performance. With high packing densities (greater than 1300 bits/in.) and multi-track operations, the problems of static and dynamic skew with head and tracking alignment problems can become severe. Particular problems occur in the following areas.

Data Capacity. The required data capacity of 9×10^8 bits per television recorder has never been achieved in a flight qualified spacecraft recorder. The RCA manufactured tape recorder used successfully on the OGO program has a storage capacity of 43×10^6 bits using nine tracks. RCA is also manufacturing a recorder (HDRSS) for the Nimbus B program which stores 56×10^6 bits on two tracks. Raymond Engineering Laboratories have flown a recorder (Model 2008) on the Giannini No. 2 program which has 60×10^6 bits on eight tracks. In development presently for the Apollo EM/I program, by Raymond, is a recorder that is expected to store 10^8 bits on 18 tracks. This represents the largest data storage in a tape recorder in development for spacecraft use and is an order of magnitude smaller than that required by the Voyager TV system.

Tape Length. To achieve the data storage required, a tape length of approximately 3000 feet will be required. Lockheed Electronics uses 2520 feet of tape on a recorder for the ISIS program and 3600 feet on the 417H program. This long length of tape required affects reliability as noted in above comments.

Data Rate. The required 696 kb/sec input data rate has never been achieved in a flight qualified spacecraft recorder. The Raymond recorder used on Mariner C had a 10.7 kb/sec input rate. Borg-Warner Model R-150 used on the Douglas Saturn S4B program had an 18 kb/sec input rate. Higher input bit rates than these can easily be achieved but generally have not been required in the past. Leach Corporation has flown a one MHz analog recorder on an LMSC program; but due to a spacecraft failure, the recorder performance could not be evaluated. Leach claims that on the basis of its testing of the one MHz analog recorder, a one megabit/sec digital rate can be achieved.

Packing Density. The packing density on Lockheeds's recorder for Mariner 69 is 1350 bpi/track. Raymond is proposing 1500 bpi/track for the Saturn Program (REL Model 2058). Numerous manufacturers have achieved approximately 1000 bpi/track on flight units. The packing density of 2330 bpi/track required for the hypothetical television system represents peak state-of-the-art performance for spacecraft recorders.

Multiple Record and Reproduce Speeds. The required two record and four reproduce speeds on a single recorder has not been achieved in a spacecraft. The use of two separate speeds, generally one for record and one for reproduce, is most common. Lockheed has used two record and a separate reproduce speed on the OWL program. Raymond has also used two record and a separate reproduce speed on the Giannini No. 1 program.

Synchronous Readout with Telemetry. This has not been accomplished before and represents a major development effort to insure satisfactory performance.

Track Switching. This is not considered a major problem and has been accomplished on other programs; however, it adds complexity and design problems to the development of the recorder.

Multiple Data Channels. The use of 14 tracks on 1/2-inch tape is not considered a major problem but does represent about the maximum that can be presently achieved.

Summary of Data Storage Problem. As discussed in Volumes 5 and 11, the data storage problem largely arises from the desire to make the performance of the hypothetical (all TV) and recommended (film and low resolution TV) science payloads comparable and at a level that provides a significant data payoff. Another way of looking at the problem is to recognize that, if state-of-the-art recorders are used, the performance of the TV system is much less than that of the film system.

1.4 TELEMETRY LIST

The telemetry list, Table 1-5, shows the measurement requirements for the hypothetical spacecraft with the all-TV photo-imaging subsystem.

The data to be transmitted falls into the following four categories:

- Measurements required for the performance of flight operations. These include measurements for diagnostic purposes to select either commanded alternate modes of operation or a degraded mission.
- Measurements required to verify the performance of specific spacecraft and subsystem functions.

- Measurements to indicate the effect of the space environment on the spacecraft performance.
- Data to evaluate critical components, in particular those newly developed for Voyager.

The categories above are listed in order of priority, and the letters assigned are those used in the list.

The channel requirements are shown by subsystem and by unit of that subsystem. Redundant units are denoted by (2) placed by the units. The function, i.e. temperature, pressure, etc. is listed in the adjacent column with the quantity of monitored points in each unit. The mode column is applicable to spacecraft engineering data but not science data as the latter has only one mode. Mode 1 is the data format used during cruise and Mode 2 is used during pre-maneuver and maneuver.

Data is of the following types: A - Analog, B - Digital, D - Discrete. The time between samples for the spacecraft data in both modes is 0.75 second for main frame words and 48 seconds for sub-commutator words. For the science commutator the corresponding times are 0.02 and 0.32 seconds.

The column headed measurements indicates the number of measurements of that type whether they are analog, binary or discretes. Analog measurements are encoded into the standard telemetry word (either 6 or 8 bits depending on whether it is engineering or science data respectively). The binary data which may consist of digital words up to 20 bits per word are sectioned into the standard telemetry word length for that channel. The discrete measurements are formatted to one measurement per bit in the telemetry word, that is, six discrete inputs to each six bit word.

Table 1-5. Telemetry List

Subsystem	Unit	Function	Mode	Type	Time Between Samples (sec)	Measurements	Priority
Guidance and control	Gyro (2)	Pitch gyro output-rate -position	1, 2	A	0.7	2	a
		Roll gyro output-rate -position	1, 2	A	0.7	2	a
		Yaw gyro output-rate -position	1, 2	A	0.7	2	a
		Pitch control signal	1, 2	A	0.7*	1	b
		Roll control signal	1, 2	A	0.7*	1	b
		Yaw control signal	1, 2	A	0.7*	1	b
	Electronics	Pitch switching amplifier output(2)	1, 2	D	0.7	2	b
		Roll switching amplifier output(2)	1, 2	D	0.7	2	b
		Yaw switching amplifier output(2)	1, 2	D	0.7	2	b
		Roll integrator	1, 2	D	48	1	b
		Mode control monitor(4)	1, 2	D	48	4	a
		Gyro torquer current generator output	1, 2	D	48	1	b
	Fine sun sensor(2) Coarse sun sensor	Sun sensor pitch output -fine -coarse	1, 2	A	0.7	2	a
	Fine sun sensor(2) Coarse sun sensor	Sun sensor yaw output -fine -coarse	1, 2	A	0.7	2	a
	Electronics	Sun Sensor acquisition gate	1, 2	D	48	1	b
	Canopus Sensor (2)	Canopus sensor roll error output	1, 2	A	0.7	1	a
		Canopus sensor star intensity output	1, 2	A	0.7	1	a
		Canopus sensor sun present signal	1, 2	D	48	1	b
		Canopus sensor star present signal	1, 2	D	48	1	b
		Canopus sensor tracking position	1, 2	D	48	2	b
	Accelerometer Electronics (2)	Accelerometer output	1, 2	D	0.7	2	b
		No. 1 thrust vector control actuator position	1, 2	A	0.7*	1	b
		No. 2 thrust vector control actuator position	1, 2	A	0.7*	1	b
		No. 1 thrust vector control actuator control signal	1, 2	A	0.7*	1	b
		No. 2 thrust vector control actuator control signal	1, 2	A	0.7*	1	b
	Reaction control system	System A gas bottle pressure	1, 2	A	48	1	a
		System B gas bottle pressure	1, 2	A	48	1	a
		System A regulated gas pressure -high	1, 2	A	48	1	b
	Reaction control system	System B regulated gas pressure -high	1, 2	A	48	1	b
		Control jet actuation (12)	1, 2	D	0.7	12	b
	Gyro (2)	Pitch gyro spin motor rotation detector	1, 2	D	48	2	b
		Roll gyro spin motor rotation detector	1, 2	D	48	2	b
		Yaw gyro spin motor rotation detector	1, 2	D	48	2	b
	Electronics	800 Hz inverter output	1, 2	A	48	1	b
		DC voltage 1	1, 2	A	48	1	b
		DC voltage 2	1, 2	A	48	1	b
		DC voltage 3	1, 2	A	48	1	b
		High-gain antenna axis position (12 bits)	1, 2	B	48	2	b
		High-gain antenna shaft axis position (12 bits)	1, 2	B	48	2	b
		Medium-gain antenna hinge axis position (12 bits)	1, 2	B	48	2	b
	Antennas	High-gain hinge actuator power	1, 2	D	48	1	b
		High-gain shaft actuator power	1, 2	D	48	1	b
		Medium-gain hinge actuator power	1, 2	D	48	1	b
	Limb and terminator sensor	Limb and terminator crossing sensor outputs (4)	1, 2	D	48	4	b

*Sampled twice in Mode 1, reducing time between samples to 0.3 second

Table 1-5. Telemetry List (Continued)

Subsystem	Unit	Function	Mode	Type	Time Between Samples (sec)	Measurements	Priority
Guidance and control (continued)	Fine sun sensors (2)	Fine sun sensor temperature	1, 2	A	48	2	c
	Coarse sun sensor	Coarse sun sensor temperature	1, 2	A	48	1	c
	Canopus sensor (2)	Canopus sensor temperature	1, 2	A	48	2	c
	Gyro (2)	Pitch gyro temperature	1, 2	A	48	2	c
		Roll gyro temperature	1, 2	A	48	2	c
		Yaw gyro temperature	1, 2	A	48	2	c
	Accelerometer (2)	Accelerometer temperature	1, 2	A	48	2	c
	Electronics	No. 1 thrust vector control actuator temperature	1, 2	A	48	1	c
		No. 2 thrust vector control actuator temperature	1, 2	A	48	1	c
	Reaction control system	System A gas bottle temperature	1, 2	A	48	1	c
		System B gas bottle temperature	1, 2	A	48	1	c
	Antenna	High-gain antenna actuator temperature	1, 2	A	48	1	c
		Medium-gain antenna actuator temperature	1, 2	A	48	1	c
	Planetary scan platform	Planetary scan platform shaft position (12 bits)	2	A	48	2	b
	Electronics	Pitch control C-1 engine on/off	1, 2	D	48	1	b
		Yaw control C-1 engine on/off	1, 2	D	48	1	b
		C-1 engine actuator (4)	1, 2	D	0.7**	4	b
Command and sequencing	Command and sequencing (2)	Temperature	2	A	48	2	c&d
		Voltage (3)	2	A	48	6	c&d
		Command events (7 bits)	1, 2	B	48	4	a
		Status (5 bits)	1, 2	B	48	2	b
		Mission time (26 bits)	1, 2	B	48	5	a
		Current	2	A	48	2	c&d
	Command (2)	Voltage (3)	2	A	48	6	c&d
		Temperature	2	A	48	2	c&d
		Command events (7 bits)	1, 2	B	48	4	a
		Verification (5 bits)	1, 2	B	0.7	4	a
		Status (5 bits)	1, 2	B	48	2	b
		Bit synchronous lock (2 bits)	1, 2	B	48	1	a
Communication	Baseband assembly	Baseband output	2	A	48	1	a
		Normal/redundant	2	D	48	2	b
		Voltage (2)	2	A	48	2	c
	Exciter (2)	Modulation	2	A	48	2	b
		Power output	2	A	48	2	a
		Voltage (3)	2	A	48	6	c
	1 w transmitter	Modulation	2	A	48	1	b
		Power output	2	A	48	1	a
		Voltage (3)	2	A	48	3	c
	Power amplifier	Output power	2	A	48	2	c
		Helix current	2	A	48	2	b
	Power supply (2)	Temperature	2	A	48	2	c
		Voltage (2)	2	A	48	4	c
	Transmitter selector	Mode (9)	2	D	48	9	a
		Voltage (2)	2	A	48	2	c
	Receiver (4)	In-lock	1, 2	D	48	4	a
		Loop stress	2	A	0.7	4	b
		Voltage control oscillator temperature	2	A	48	4	c
		Voltage (2)	2	A	48	8	c
	Receiver selector	Mode (5)	2	D	48	5	a
		Voltage (2)	2	A	48	2	c
	Low-gain antenna selector	Mode (3)	2	D	48	3	a
		Voltage (2)	2	A	48	2	c
	Circulator assembly	Mode (5)	2	D	48	5	a
		Voltage (2)	2	A	48	2	c
	Power supply	Input voltage	2	A	48	1	b
		Output voltage (5)	2	A	48	5	a
	Low data rate	Voltage	2	A	48	1	c
	Receiver demodulator capsule relay	Signal present temperature	2	D	48	1	a
		Temperature	2	A	48	1	c
	High data rate	Voltage	2	A	48	1	c
	Receiver demodulator capsule relay	Signal present	2	D	48	1	a
		Temperature	2	A	48	1	c

** Sampled four times in Mode 1, reducing time between samples to 0.08 second.

Table 1-5. Telemetry List (Continued)

Subsystem	Unit	Function	Mode	Type	Time Between Samples (sec)	Measurements	Priority
Communication (continued)	UHF receiver capsule relay	Signal strength	2	A	48	1	b
		Temperature	2	A	48	1	c
		Voltage	2	A	48	1	c
Telemetry and data storage	Telemetry digital unit (2)	Calibration	2	A	48	4	c
		Voltage (2)	2				
		Temperature	2	A	48	2	d
	Telemetry analog unit (2)	Voltage (3)	2	A	48	6	c
	Recorder (6)*	Tape in motion	2	D	48	6	a
		Read mode	2	D	48	6	a
		Write mode	2	D	48	6	a
		End of tape	2	D	48	6	a
		Temperature	2	A	48	6	c
		Pressure	2	A	48	6	c
Thermal control	Structure	Temperature (40)	2	A	48	40	b
Capsule	Capsule	Undefined (5)	2	A	48	5	b
		Undefined (5)	2	D	48	5	c
Power	Power control unit	Oscillator voltage	2	A	48	1	b
		Voltage (6)	1, 2	A	48	6	a
		Current (2)	2	A	48	2	c&d
		Oscillator main-standby	2	D	48	1	c&d
		Temperature (2)	2	A	48	2	b
		Status signals (10)	2	D	48	10	b
	Battery (3)	Voltage	2	A	48	3	a
		Current (2)	1, 2	A	48	6	a
		Current sign	1, 2	D	48	3	a
		Temperature (2)	2	A	48	6	b
	400-Hz inverter	Voltage (2)	1, 2	A	48	2	a
		Current input	2	A	48	1	b
		Current output (2)	2	A	48	2	a
		Inverter main-standby	2	D	48	1	c&d
	Solar array	Current (4)	2	A	48	4	a
		Temperature (8)	2	A	48	8	b
		Voltage (2)	2	A	48	2	a
Propulsion	Pressurization system	Helium tank pressure	1, 2	A	48	1	b
		Fuel tank inlet pressure	1, 2	A	48	1	b
		Regulator discharge pressure	1, 2	A	48	1	b
		Oxidizer inlet pressure	1, 2	A	48	1	b
	Feed system	Oxidizer tank discharge pressure	1, 2	A	48	1	b
		Fuel tank discharge pressure	1, 2	A	48	1	b
		Fuel tank inlet pressure	1, 2	A	48	1	b
		Oxidizer start tank discharge pressure	1, 2	A	48	1	b
	Feed system (continued)	Fuel start tank discharge pressure	1, 2	A	48	1	b
	LM engine	Engine feed system interface oxidizer pressure	1, 2	A	0. 7**	1	a
		Engine feed system interface fuel pressure	1, 2	A	0. 7**	1	a
		Injector inlet oxidizer pressure	1, 2	A	48	1	a
		Injector inlet fuel pressure	1, 2	A	48	1	a
		LM chamber pressure (2)	1, 2	A	0. 7**	2	a
	Pressurization system	Helium tank temperature (2)	1, 2	A	48	2	b
	Feed system	Main oxidizer propellant tank temperature	1, 2	A	48	2	b
		Main fuel propellant tank temperature	1, 2	A	48	2	b
		Oxidizer propellant inlet to engine temperature	1, 2	A	48	1	b
		Fuel propellant inlet to engine temperature	1, 2	A	48	1	b
	Engine (thrust chamber)	Skirt temperature (2)	1, 2	A	48	2	b
		Outer skirt temperature (2)	1, 2	A	48	2	b
		Head end temperature (2)	1, 2	A	48	2	b
		Combination chamber temperature (3)	1, 2	A	48	3	b
	Engine (thrust chamber)	Throat temperature (2)	1, 2	A	48	2	b
		Actuator temperature	1, 2	A	48	1	b

* In film/TV system only four recorders are used with a corresponding reduction of channels.

**These parameters occupy 2 words in Mode 1, reducing time between samples to 0. 3 second.

Table 1-5. Telemetry List (Continued)

Subsystem	Unit	Function	Mode	Type	Time Between Samples (sec)	Measurements	Priority
Propulsion (continued)	Pressurization system	Explosive valve actuation position	1, 2	D	48	15	a
		Tank vent pressure switch actuation (position	1, 2	D	48	1	a
	LM engine	Pintle actuator current	1, 2	A	48	1	b
		Pintle actuator position	1, 2	D	48	1	a
		Quad solenoid valve current (2)	1, 2	A	48	2	b
		Quad solenoid valve position (4)	1, 2	D	48	16	a
		Pre-valve position	1, 2	D	48	8	a
		Pre-valve current	1, 2	A	48	1	b
		Pilot operated ball valve position	1, 2	D	48	8	a
		Pilot operated ball valve current	1, 2	A	48	2	b
	C-1 engine (4)	C-1 engine pressure	1, 2	A	0.7*	4	b
Electrical distribution	Distribution control unit	Distribution power supply voltage	1, 2	A	48	1	a
		Distribution power supply temperature	2	A	48	1	c
		Low-thrust engine valve switch on/off A	1, 2	D	48	1	a
	Distribution control unit (continued)	Low-thrust engine valve switch on/off B	1, 2	D	48	1	a
		High-thrust engine valve switch on/off A	1, 2	D	48	1	a
		High-thrust engine valve switch on/off B	1, 2	D	48	1	a
		Start tank engine valve switch on/off A	1, 2	D	0.7	1	a
		Start tank engine valve switch on/off B	1, 2	D	0.7	1	a
		Pre-valve switch on/off A	1, 2	D	0.7	1	a
		Pre-valve switch on/off B	1, 2	D	0.7	1	a
		Thrust vector controls power on/off A	1, 2	D	48	1	a
		Thrust vector controls power on/off B	1, 2	D	48	1	a
		High gain antenna hinge motor drive heater on/off A	2	D	48	1	b
		High gain antenna hinge motor drive heater on/off B	2	D	48	1	b
		High gain antenna shaft motor drive heater on/off A	2	D	48	1	b
		High gain antenna shaft motor drive heater on/off B	2	D	48	1	b
	Distribution control unit (continued)	Medium gain antenna hinge motor drive heater on/off A	2	D	48	1	b
		Medium gain antenna hinge motor drive heater on/off B	2	D	48	1	b
		Planetary spin platform gimbal motor heater on/off A	2	D	48	1	b
		Planetary spin platform gimbal motor heater on/off B	2	D	48	1	b
		Command subsystem power on/off A	1, 2	D	48	1	a
		Command subsystem power on/off B	1, 2	D	48	1	a
		S-band radio subsystem power on/off A	1, 2	D	48	1	a
		S-band radio subsystem power on/off B	1, 2	D	48	1	a
		Control and sequencer subsystem power on/off A	1, 2	D	48	1	a
		Control and sequencer subsystem power on/off B	1, 2	D	48	1	a
		Guidance and control subsystem power on/off A	1, 2	D	48	1	a
		Guidance and control subsystem power on/off B	1, 2	D	48	1	a

*These parameters occupy 2 words in Mode 1, reducing time between samples to 0.3 second.

Table 1-5. Telemetry List (Continued)

Subsystem	Unit	Function	Mode*	Type	Time Between Samples (sec)	Measurements	Priority
Electrical distribution (continued)	Distribution control unit (continued)	Propulsion subsystem power on/off A	1, 2	D	48	1	a
		Propulsion subsystem power on/off B	1, 2	D	48	1	a
		Telemetry and data subsystem power on/off A	1, 2	D	48	1	a
		Telemetry and data subsystem power on/off B	1, 2	D	48	1	a
		Capsule power on/off A	1, 2	D	48	1	a
		Capsule power on/off B	1, 2	D	48	1	a
		Reaction control system heater power on/off A	1, 2	D	48	1	a
		Reaction control system heater power on/off B	1, 2	D	48	1	a
		Experiment power on/off A	1, 2	D	48	1	a
		Experiment power on/off B	1, 2	D	48	1	a
		UHF radio link power on/off A	1, 2	D	48	1	a
		UHF radio link power on/off B	1, 2	D	48	1	a
	Pyro control unit	Pyro power supply voltage	1, 2	A	48	1	a
		Pyro power supply temperature	2	A	48	1	c
		Pre-arm	1, 2	D	48	1	a
		Safe/Arm 1 A	1, 2	D	48	1	a
		Safe/Arm 1 B	1, 2	D	48	1	a
		Safe/Arm 2 A	1, 2	D	48	1	a
		Safe/Arm 2 B	1, 2	D	48	1	a
	Pyro control unit (continued)	Safe/Arm 3 A	1, 2	D	48	1	a
		Safe/Arm 3 B	1, 2	D	48	1	a
		Safe/Arm 4 A	1, 2	D	48	1	a
		Safe/Arm 4 B	1, 2	D	48	1	a
		Safe/Arm 5 A	1, 2	D	48	1	a
		Safe/Arm 5 B	1, 2	D	48	1	a
Structure	Separation switches (4)	Separation	2	D	48	4	b
	Low gain antennas (2)	Low gain antenna extended	2	D	48	2	b
	Medium gain antenna	Medium gain antenna released	2	D	48	1	b
	High gain antenna	High gain antenna released	2	D	48	1	b
	Planetary scan platform	Planetary scan platform released (intermediate) position)	2	D	48	1	b
		Planetary scan platform released (full position)	2	D	48	1	b
		Planetary scan platform uncaged	2	D	48	1	b
Science	H/R infrared spectrometer	Multiplexer output (8 bits)		B	0.02	1	b
		Grating position		A	0.32	1	b
		Internal multiplexer status		A	0.32	1	b
		PbSe detector temperature		A	0.32	1	c
		Calibration monitor		A	0.32	1	b
		Bias voltage level monitor		A	0.32	1	b
		AGC status monitor		A	0.32	1	b
		Power supply voltage monitor		A	0.32	1	b
		Electronics temperature		A	0.32	1	c
	Broadband infrared spectrometer	Multiplexer output (8 bits)		B	0.02	1**	b
		Detector No. 1 temperature		A	0.32	1	c
		Detector No. 2 temperature		A	0.32	1	c
		Chopper temperature, No. 1 and No. 2		A	0.32	2	b
		Internal multiplexer status		A	0.32	1	b
		Telescope and monochrometer temperature		A	0.32	1	c
		Electronics temperature		A	0.32	1	c
	Infrared radiometer	Radiometer output		A	0.02	1***	b
		Mirror position		A	0.32	1	b
		Electronics temperature		A	0.32	1	c
		Sensor 1 temperature		A	0.32	1	c
		Sensor 2 temperature		A	0.32	1	c
		Sensitivity status high/low		A	0.32	1	b
		Calibration monitor		A	0.32	1	c
	Ultraviolet spectrometer	UV spectrometer output		A	0.02	1***	b
		High voltage monitor No. 1 and No. 2		A	0.32	2	b
		Power supply temperature No. 1 and No. 2		A	0.32	2	c
		Grating position monitor		A	0.32	1	b
		Electronics temperature		A	0.32	1	c
		Multiplexer status		A	0.32	1	b

*Science commutator does not have more than one mode
**Repeated 4 times
***Repeated 6 times

Table 1-5. Telemetry List (Continued)

Subsystem	Unit	Function	Mode*	Type	Time Between Samples (sec)*	Measurements	Priority
Science (Continued)	Computer and Sequencer	Mission time (26 bits)		A	0.32	4	a
	3 medium-resolution TV and 1 high-resolution TV	Target temperature		A		4	c
		Heater current		A		4	b
		High voltage		A		4	b
		High light level		A		4	b
		Shutter position		A		4	b
		Power transformer temperature		A		4	c
		Power supply voltages (6)**(8)***		A		26	b
		Timing reference (18 bits)		B		12	b
		Power-on command		D		4	b
		Shutter command		D		4	b
		Scan/record command		D		4	b
		Erase command		D		4	b
		Focus command****		D		3	b
		Trio command		D		4	b
		Calibrate command		D		4	b
	Low-resolution color TV	Target temperature		A		1	c
		Heater current		A		1	b
		High voltage		A		1	b
		High light level		A		1	b
		Shutter position		A		1	b
		Filter wheel position		A		1	b
		Power supply voltages (6)		A		6	b
		Time reference (18 bits)		B		3	b
		Power-on command		D		1	b
		Shutter command		D		1	b
		Scan/record command		D		1	b
		Erase command		D		1	b
		Trio command		D		1	b
		Calibrate command		D		1	b
	Film system	Total film system		A		24	-
		Total film system		B		5	-
		Total film system		D		7	-

*Photo-imaging multiplexer does not have more than one mode, and time between samples is not applicable.
**Low- and medium-resolution TV cameras require six power supply TM signals.
***High-resolution TV cameras require eight power supply TM signals.
****Not used on high-resolution camera.

2. S-BAND SUBSYSTEM

2.1 SUMMARY

The S-band radio subsystem provides the data link for communicating between the spacecraft and the ground tracking stations of the Deep Space Network. The subsystem (Figure 2-1) includes the S-band antennas, the receiving and transmitting equipment, and related control and monitoring circuitry. A simplified block diagram of the recommended subsystem configuration together with preliminary specifications is shown in Figure Figure 2-2.

During six months in Martian orbit the communication link will support an average data rate of 15 to 40 kb/sec; the actual value depends on the trajectory and the actual performance of the communication system (see also Table 2-6). Simultaneously, it provides a continuous 512 bit/sec link for transmission of the engineering and lower rate science data.

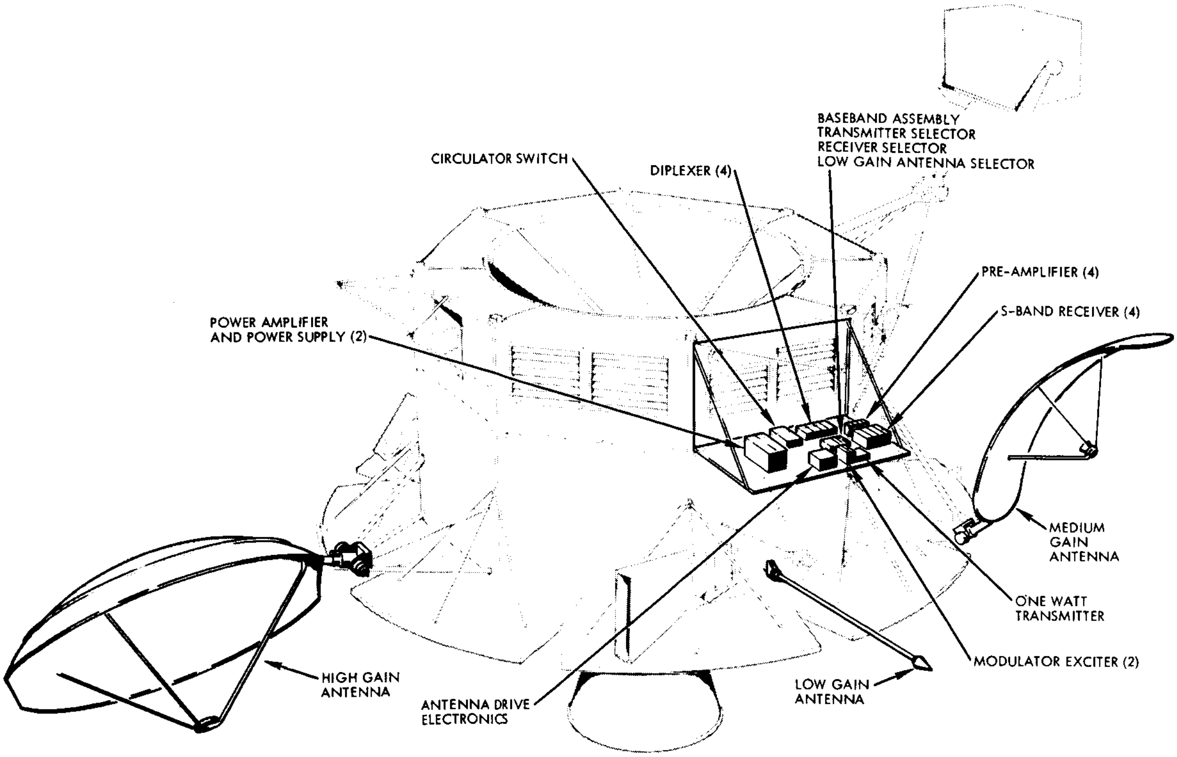


Figure 2-1

THE S-BAND RADIO SUBSYSTEM includes two low gain antennas, a medium gain antenna and a high gain antenna. The electronic equipment for this subsystem is mounted on a single panel of the equipment module.

PRELIMINARY SPECIFICATION
S—Band Radio Subsystem

Purpose

The S-band radio subsystem provides reception and demodulation of command and ranging data from the DSN ground stations and transmission of telemetry data and ranging from the spacecraft to the ground station. The subsystem also provides for a doppler frequency shift measurement by transmitting a carrier frequency coherently related to the received uplink frequency.

Physical Characteristics

Weight:
- S-band electronics 68.6 pounds
- Antennas and drives 135.6 pounds

Power:
- S-band electronics 180 watts max
- Antennas and drives 60 watts peak

Performance Characteristics

RECEPTION:
- Number of channels: Four (2 low gain, 1 medium gain and 1 high gain)
- Frequency: 2110 - 2120 MHz
- Receiver sensitivity: -149 dbm
- Noise figure: 5 db maximum
- Phase lock bandwidth: 32 Hz
- IF Bandwidth
  - wideband channel: 2.5 MHz
  - narrowband channel: 1 KHz
- Receiver dynamic range: -149 to -50 dbm

TRANSMISSION:
- Number of transmission channels: three (1 low gain, 1 medium gain and 1 high gain)
- Frequency: 2290 to 2300 MHz (coherent with uplink or optional internal crystal control)
- High power transmitter output: 50 watts minimum
- Low power transmitter output: 1 watt minimum
- Transmission bandwidth: 3 MHz
- Number of data subcarrier channels: three
- High rate channel bandwidth: 800 KHz (ranging or telemetry)
- Engineering data channel bandwidth: 10 KHz centered at 1.024 MHz
- Low rate data channel: 200 Hz centered at 2.73 KHz

ANTENNAS:
- High Gain
  - Type: double gimballed circular paraboloid
  - Gain: 34 db
  - 3 db beamwidth: 3.2 degrees
- Medium Gain
  - Type: single gimballed elliptical paraboloid
  - Gain: 28 db
  - 3 db beamwidth: 4 degrees by 10 degrees
- Low Gain
  - Type: four arm conical spiral
  - Gain: +1 db
  - 3 db beamwidth: 180° minimum

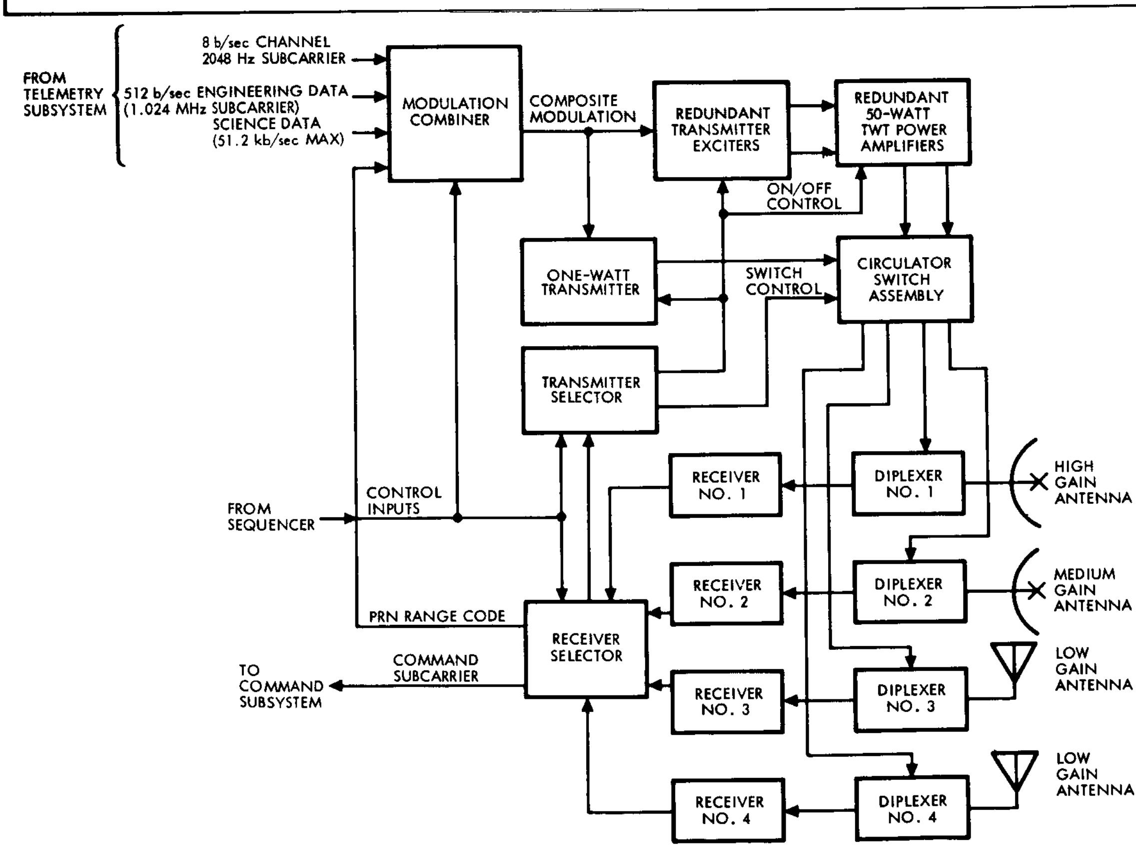


Figure 2-2

The link will support 8 bit/sec command data through omnidirectional antennas. Pseudo random noise ranging is provided through the entire mission over high or medium-gain antenna channels.

Four separate S-band antennas are provided. A two-axis steerable high gain antenna provides 34 db of gain at a beamwidth of 3.2 degrees for normal cruise and orbital operations. A second directional antenna provides a fan-shaped beam approximately 4 x 10 degrees, steerable in one axis. This antenna has a nominal gain of 28 db and is used for pre-maneuver operations and as a backup to the high gain antenna for orbital operations. Two low gain antennas, one located on each side of the spacecraft, give spherical coverage for maneuver sequences and emergency (unoriented) communications. Each of the four antennas is connected through a diplexer which provides frequency-selective filtering to permit simultaneous transmission and reception.

The receiver channel output of each diplexer is connected to a separate phase-locked receiver which synchronously demodulates the uplink data and provides a coherently related output carrier frequency to drive the downlink transmitters. Each receiver provides a narrowband output signal containing the command data subcarrier and a wideband output containing the pseudorandom noise ranging signal. These outputs are processed through logic circuitry which selects the best receiver channel. Selection is based on a combination of command-controlled priority logic and relative signal to noise ratio. Normally the highest received signal to noise ratio is selected.

For the 8 bit/sec command rate, a bit synchronizer that establishes synchronization on the data itself is recommended (see Section 3) rather than one that requires power for a separate synchronizing signal. Comparing the self-synchronizing 8 bit/sec system with a 1 bit/sec system containing a separate sync signal, such as the system flown on Mariner C, indicates that only 0.6 db additional power is required to achieve the factor of eight increase in data rate. An additional benefit derived from the higher command rate is that acquisition time is dramatically reduced. In the self-synchronizing system, acquisition takes less than two minutes, while for the Mariner C type system, acquisition requires in excess of 20 minutes.

The recommended S-band subsystem is compatible with any command rate between 8 and 128 bits/sec.

The downlink consists of modulation-combining equipment, two redundant 50-watt transmitter channels, and one low-power transmitter channel. The transmitter outputs are routed to a circulator switch assembly which permits (under command or computer and sequencer control) any of the three transmitter outputs to be connected to any of the four antennas through the appropriate diplexer.

Essentially, the downlink provides three channels for transmission of data from the spacecraft to the ground station. The high-rate channel directly phase-modulates the carrier and provides a baseband transmission bandwidth up to 800 kHz. This channel is time shared between the ranging code and the science and video mapping data. The second channel provides 512 bit/sec engineering telemetry and low rate science data on a 1.024 MHz subcarrier. The third channel provides 8 bit/sec telemetry data on a 2.048 kHz subcarrier. This channel is normally used only when the vehicle is performing a reorientation maneuver or when attitude stablilization has been lost.

2.2 SUBSYSTEM DESCRIPTION AND PERFORMANCE

Each of the four S-band antennas is connected permanently to one receiver via a diplexer, but any transmitter can be connected to any antenna by means of the circulator switch assembly (Figure 2-2). The output of each receiver passes to a receiver selector unit that transfers one set of output signals to other spacecraft subsystems. Normally the receiver that is connected to the same antenna being used for transmission is selected. If this receiver is not in lock, the other receivers are used in turn, on a relative signal strength basis.

The telemetry and range-code channels in operation are linearly combined in the modulation combiner, and the resultant composite signal is phase-modulated onto one of the three transmitter carriers. Each 50-watt transmitter includes a 100 milliwatt exciter driving a 50-watt traveling-wave-tube amplifier. The one-watt output is derived from a solid-state transmitter similar in design to the 100-milliwatt exciters. The transmitter carrier frequency is derived from the doppler-perturbed received frequency when the uplink is locked. At other times it receives its drive from an internal crystal oscillator operating at the same nominal frequency as the receiver drive.

2.2.1 Uplink

The S-band uplink receives and demodulates command and ranging signals originating from the Deep Space Network. Because the command link is essential for achieving mission objectives and providing access in the event of a failure, its reliability is of paramount importance. This was the primary consideration in selecting two low gain antennas to provide continuous command access independent of vehicle orientation. Separate receivers for each of the low gain antennas, rather than predetection diversity combining or input-switching logic, were dictated by the need to avoid mutual interference due to spacecraft tumbling or multipath propagation effects. To provide a smooth transition between the two antennas in the presence of tumbling or antenna pattern anomalies, post-detection diversity combining of the command subcarriers is employed. Mutual interference is avoided by incorporating gating circuitry to inhibit a receiver channel output if it falls below a predetermined threshold signal to noise ratio. Reception over the high gain or medium gain antenna channels is not actually required for command operation. The primary function of these channels is to receive and demodulate the pseudorandom noise ranging data. They do, however, provide redundancy as well as additional command margin and will therefore be used for normal mission operations.

The choice of 8 bits/sec for the command transmission rate was dictated primarily by two factors. The first factor is that for a reasonably efficient modulation technique, the system will support 8 bits/sec out to the worst case communication distance (approximately 395×10^6 km). The second factor is that for data rates of 8 bits/sec and higher, it is feasible to construct bit synchronizers which will establish synchronization on the data itself, rather than requiring power for a separate synchronizing signal.

A command link power budget showing carrier and command data performance margins is given in Table 2-1. The calculations are based on the spacecraft low gain antenna and the 210 foot Deep Space Network antenna

Table 2-1. Uplink Power Budget*

Mode: Spacecraft Low Gain Antenna, 210-Foot DSN Antenna

Communication Distance: 395 x 10^6 Kilometers (Encounter Plus 6 Months)

No.	Parameter	Value	Tolerance +	Tolerance -
1	Transmitter Power	+80.8 dbm	0.0	-
2	Transmitting Circuit Loss	-0.4 db	0.1	0.1
3	Transmitting Antenna Gain	+60.6 db	1.0	0.5
4	Transmitting Antenna Pointing Loss	0.5 db	-	-
5	Space Loss 2115 MHz at 340 x 10^6 km	-271.0 db	0.0	0.0
6	Polarization Loss	-0.5 db	-	-
7	Receiving Antenna Gain**	-2.0 db	2.0	0.0
8	Receiving Antenna Pointing Loss	-0.5 db	-	-
9	Receiving Circuit Loss	-1.3 db	0.2	0.2
10	Total Received Power	-135.6 dbm	3.3	0.8
11	Receiver Noise Spectral Density T System = 930 deg	-168.9 dbm/Hz	1.0	1.0
Carrier Performance				
12	Modulation Loss	-3.2 db	0.5	0.5
13	Received Carrier Power	-138.5 dbm	3.8	1.3
14	Received Carrier SNR ($2B_L$ = 30 Hz)	15.1 db	4.8	2.3
15	Required Carrier SNR	6.0 db	-	-
16	Margin	9.1 db	4.8	2.3
Command Performance (8 bit/sec)				
17	Modulation Loss	-2.9 db	0.5	0.5
18	Demodulator Degradation	-1.0 db	0.0	0.5
19	Received Subcarrier Power	-138.5 dbm	3.8	1.3
20	Received SNR (8 bit/sec)	20.4 db	4.8	2.8
21	Required SNR (1 x 10^{-6} error rate)	10.5 db	-	-
22	Margin	9.9 db	4.8	2.8

*Uplink power budget shows carrier and command data performance margins at the maximum communication distance of 395 x 10^6 kilometers. Calculations show that the above expected performance exceeds substantially the negative tolerance on the estimate.

**Worst-case look angle.

with a 100 kw transmitter. Use of the 210 foot antenna is based on the premise that the simultaneous loss of spacecraft orientation and unavailability of a 210 foot antenna will not occur. The link margins available at encounter plus six months are 9.1 db for the carrier and 9.9 db for the command data based on a 10^{-6} error rate.

The 8 bit/sec data is biphase modulated onto a subcarrier, which in turn phase modulates the main carrier. The subcarrier frequency is chosen based on a compromise between having the frequency too low so that the carrier phase-lock loop would interfere, and having it too high with an attendant increase in the dynamic range required of the subcarrier demodulator to offset predetection suppression in the receiver limiting intermediate frequency amplifiers.

Each of the four receivers provides a DC output signal proportional to its received carrier amplitude. Provided this signal exceeds a value proportional to the allowable command error rate threshold, the uplink channel corresponding to the selected downlink antenna will be selected. In the event the priority channel is below threshold, the channel having the highest signal to noise ratio is used. For example, during the normal cruise phase and during orbital operations, the high gain antenna will be selected. If the receiver connected to that antenna drops below threshold due to antenna failure or a loss of orientation, the selection logic will then select the medium gain antenna provided it is being illuminated by the earth station. If not, it will select the diversity combined output from the low gain channels. The low gain antenna selection signal is routed to the transmitter selector so that the correct antenna will be selected when transmitting in the low gain mode. Provisions will be included for ground controlled and failure-sensing override of the primary selection logic.

For normal mission support operations, the uplink reception mode will depend on the particular phase of the mission. These reception modes are summarized in Table 2-2. During the early cruise phase prior to sun and Canopus acquisition, reception will be via the low gain antennas. Also, the low gain antenna will be used during reorientation maneuvers. For all other operational support, the high gain antenna will normally be used. The only exception to this is during a premaneuver phase when the medium

Table 2-2. Various Uplink Receiving Modes*

Antenna	Normal Use	Remarks
Low Gain (either)	Prior to initial orientation and during reorientation maneuvers	Can be used for commands during entire mission.
High Gain	Interplanetary cruise/and orbital operations	Primarily required for PRN ranging reception.
Medium Gain	Backup to high gain antenna for injection and orbital operation	May be used during pre-maneuver phase when the high gain antenna is repositioned to provide coverage during the maneuver.

*Various uplink receiving modes are available depending on the phase of the mission. Any of the receiving modes may be utilized when the spacecraft is oriented. Command reception is achieved for any spacecraft orientation throughout the mission.

gain or low gain may be used for reception while the high gain antenna is repositioned to the post-orientation maneuver angle.

The uplink spectrum occupancy is dictated primarily by the pseudo random noise ranging signal which operates at a clock frequency of 500 kHz and has a sine x/x spectrum with nulls at 1 MHz and 2 MHz giving a total RF spectrum occupancy of 4 MHz. The command subcarrier frequency of 512 Hz is sufficiently low that it does not impose any constraints on spectrum utilization. Two spacecraft consume a bandwidth of 9 MHz including a 1 MHz guard band, leaving approximately 1 MHz for the two capsule links. This appears to be more than adequate considering that range measurements to the capsule are not required. In the event more spectrum is required, the ranging signal can readily be truncated at 1 MHz without degrading of the range measurements significantly. This reduces the uplink spectrum to 5 MHz including the guard band.

2.2.2 Downlink

The downlink transmits four types of data. These include diagnostic and state-of-health engineering data during the entire mission, low rate science data during the orbital phase, high rate science and video mapping data during the orbital phase, and pseudo random noise ranging data on a periodic basis during the entire mission. The downlink transmission rates are essentially constrained by the available effective radiated power (ERP). The maximum ERP for the recommended Voyager configuration has been set at +81 dbm. This includes a 34-db spacecraft antenna gain and a +47-dbm transmitter power. The antenna gain has been constrained by the available pointing accuracy of ±0.8 degree and by physical dimensions associated with simplified antenna stowage. Larger antenna apertures would require greater pointing accuracies and also some form of feed structure deployment mechanism with an attendant reduction in reliability. Transmitter power has been constrained to 50 watts to allow adequate power margin with conveniently mounted fixed solar array panels.

The choice of a modulation structure and data rates for downlink transmission is to a large extent dependent upon specific mission requirements. The link is capable of supporting data rates in excess of 50 kb/sec at nominal encounter communication distances. The available rate decreases by about a factor of four between nominal 1973 encounter and the end of the six-month mission. To get the highest data return for this transmission-rate profile versus time, it is desirable to have adaptive data rates. During orbital operations by far the most demanding requirements for data transmission are imposed by the video mapping system. Since transmission of these data requires most of the available ERP, a modulation structure has been selected which provides maximum flexibility in bit rates for transmission of the mapping data. This is achieved by split-phase modulating the mapping data directly onto the downlink carrier. This provides a channel capable of transmitting coded data at information rates varying between 500 bits/sec and 50 kb/sec. The data rate for transmission of the engineering data has been selected at 512 bits/sec to provide a balance between reasonable sampling rates and minimum power consumption. A second 8 bit/sec mode is also provided for transmission of engineering data as a backup to the normal 512 bits/sec. The pseudo random noise ranging data, which requires a base bandwidth of 800 kHz, time-shares the high rate channel with the high rate mapping data. The selected modulation structure is compatible with both uncoded transmission or transmission using the 32, 6 biorthgonal code. A tabulation of the available data rates is given in Table 2-3.

Table 2-3. Available Telemetry Rates*

Subcarrier Channel	Data Rate (bit/sec)	32, 6 Symbol Rate (bits/sec)	Subcarrier Frequency (Hz)
Data (split phase)	51.2 K	273 K	273 K
	25.6 K	136.5 K	136.5 K
	12.8 K	68.25 K	68.25 K
	6.4 K	34.125 K	34.125 K
	3.2 K	17.063 K	17.063 K
	512	2.73 K	2.73 K
Engineering	512	2.73 K	1.024 M
Emergency	8	42.6	2.73 K

*Available telemetry rates for each of the three data link channels illustrate data rate flexibility. Symbol rate refers to the 32,6 coded transmission rate. The data channel accommodates housekeeping and low rate sicence data, and the e mergency channel may be used to provide diagnostic data.

Two factors were instrumental in selecting 1.024 MHz as the subcarrier frequency for the 512 bits/sec engineering data. The first is that the requirement to transmit simultaneously the pseudo random noise ranging and the engineering data dictates that the subcarrier be located outside the 800 kHz pseudo random noise ranging bandwidth to avoid mutual interference. The second factor is that the specific choice of 1.024 MHz makes this channel compatible with reception of data by the manned spaceflight network which can provide tracking and data acquisition support during the early cruise phase.

A typical transmission power budget for the telemetry link is given in Table 2-4. The table includes the expected values of each parameter and the tolerance on the parameter. Performance margins are shown for the carrier loop, the high-rate mapping channel, and the engineering telemetry channel. The modulation indices have been adjusted to provide a margin for each service equal or greater than the sum of the negative

Table 2-4. Downlink Power Budget*

Mode: Spacecraft High Gain Antenna, 210-Foot DSIF Antenna

Communication Distance: 190×10^6 Kilometers (Nominal Encounter)

No.	Parameter	Value	Tolerance +	Tolerance -
1	Transmitting Power	+47.0 dbm	1.0	0.0
2	Transmitting Circuit Loss	-2.1 db	0.8	0.8
3	Transmitting Antenna Gain	+34.0 db	0.3	0.5
4	Transmitting Antenna Pointing Loss	-0.4 db	0.4	0.3
5	Space Loss 2295 MHz at 190×10^6 km	-265.4 db	0.0	0.0
6	Polarization Loss	-0.1 db	-	-
7	Receiving Antenna Gain	61.7 db	1.0	0.5
8	Receiving Antenna Pointing Loss	-0.5 db	-	-
9	Receiving Circuit Loss	-0.2 db	0.1	0.1
10	Total Received Power	-126.0 dbm	3.6	2.2
11	Receiver Noise Spectral Density T System = 45 ± 10°K	-182.0 dbm/Hz	1.1	0.9
Carrier Performance				
12	Modulation Loss	7.2 db	1.7	2.2
13	Received Carrier Power	-138.4 dbm		
14	Received Carrier SNR ($2B_L$ = 12 Hz)	38.0 db		
15	Required Carrier SNR	6.0 db		
16	Margin	32.0 db	6.4	5.3
High Rate Telemetry Performance (51.2 (kb/sec)				
17	Modulation Loss θ = 1.5 ± 10 percent	-1.2 db	0.5	-0.6
18	Demodulator Degradation	-1.0 db	-	0.5
19	Received Subcarrier Power	-128.2 dbm		
20	Received SNR (51.2 kb/sec)	6.7 db		
21	Required SNR (5×10^{-3} error rate, 32/6 code)	2.0 db		
22	Margin	4.7 db	5.2 db	4.2 db (1.6 db RMS)
Engineering Telemetry Performance (512 bits/sec)				
23	Modulation Loss θ = 0.43 ± 10 percent Demodulator Degradation	-18.3 db -1.0 db	2.4 -	3.5 0.5
24	Received Subcarrier Power	-145.3 dbm		
25	Received SNR (512 bits/sec)	9.6 db		
26	Required SNR (5×10^{-3} error rate 32/6 code)	2.0 db		
27	Margin	7.6 db	7.6 db	7.1 db (3.9 db rms)

*Downlink power budget shows the link margins available for a nominal 190×10^6 kilometers Mars encounter. Link margins are calculated for the carrier channel, a 5.12 kb/sec data channel and a 512 bits/sec engineering channel. In each case, the margins are equal or greater than the sum of the negative tolerances.

tolerances. The budget is based on the spacecraft high gain antenna, the 210-foot Deep Space Network antenna, and a 190×10^6 -km nominal encounter distance.

Table 2-5 summarizes the margins available at encounter for an early, a nominal, and a late encounter date. The peak and RMS negative tolerances are also shown for comparison.

The value of having selectable bit rates is illustrated in Figure 2-3 which shows a data rate profile as a function of time for the 1973 opportunity. Three data rate profiles are shown. The highest data rate curve is based on nominal communications link performance. The lower curves are based on a degradation equivalent to the RMS and to the peak values of the negative link tolerances. Three possible arrival dates corresponding to an early, a nominal, and a late arrival are indicated on the abscissa. It is apparent that the average achievable data rate is dependent on the arrival time because of the steepness of the curves within the range of possible arrival dates.

The value of the adaptive data rates is further illustrated in Table 2-6 which shows the average data rate obtained by integrating the curves of Figure 2-3 (quantized to the available transmission rates).

Table 2-5. Communication Margins at Arrival*

	Margin at Arrival (db)			Negative Tolerance (db)	
Service	Early	Nominal	Late	Sum	RMS
Carrier	33.0	32.0	30.0	5.3	3.0
High Rate Telemetry (51.2 kb/sec)	5.7	4.7	5.3 (25.6 kb/sec)	4.2	1.6
Engineering Telemetry (512 bits/sec)	8.6	7.6	7.7	7.4	3.9

*Communication margins at arrival indicate performance potential for nominal and worst case conditions during the critical site location and capsule descent phase. For late encounter, margins are based on 25.6 (kb/sec) and a corresponding reallocation of power between the engineering and high rate channels.

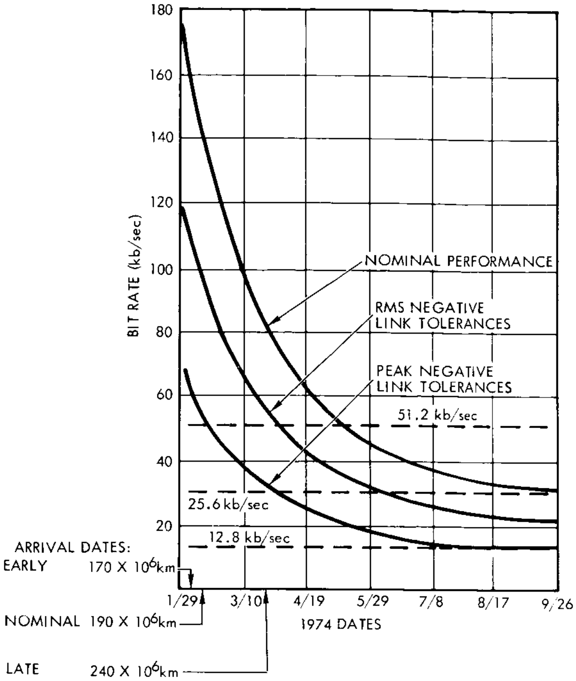


Figure 2-3

DATA RATE VERSUS TIME shows variation in data rate capability as a function of time for the 1973 mission. The three curves indicate the achievable data rates for: (1) nominal link performance, (2) the RMS of the negative link tolerances, and (3) the sum of the negative link tolerances. The horizontal lines indicate the available transmission rate as a function of time. The communication distances for an early, a nominal and a late arrival at Mars are shown on the abscissa.

Table 2-6. Average Data Rates*

Arrival Date	Communication Distance (kilometers)	Average Data Rate for 6-Month Orbital Life (kbits/sec): Nominal Performance	RMS Negative Tolerance Degradation	Peak Negative Tolerance Degradation
2 Feb 1974	170×10^6	40	32	20
14 Feb 1974	190×10^6	38	29	18
20 Mar 1974	240×10^6	33	22	15

*Average data rates for the 1973 mission depend upon the arrival date and on actual communications performance obtained. Average data rate may vary by nearly three to one between an early launch with nominal link performance and a late launch with link degradation equivalent to the sum of the negative tolerances. The average rates are derived from the data rate versus time shown in Figure 2-3 using the selected data rates of 51.2, 25.6, and 12.8 kb/sec.

The average data rate achieved varies from 40 kb/sec for an early encounter and nominal link performance to 15 kb/sec for late encounter and a link degradation corresponding to the peak value of the negative tolerances. It should be noted that the link is perfectly adaptive in the sense that the data rate need not actually be switched until the ground received error rate exceeds an acceptable level. On this basis the most probable average data rate for 6 months in orbit will be in the neighborhood of 30 kb/sec.

The 8 bit/sec channel is normally required only during reorientation maneuvers when the high and medium gain antennas are not pointed earthwards. A power budget for the 8 bit/sec channel is given in Table 2-7 for a 240×10^6 - km encounter distance (late arrival). The budget indicates that the data can just be received at late encounter with the low gain antenna and maximum adverse link tolerances. Efforts will be made to improve the margins in this mode by incorporating aided tracking to reduce the downlink carrier power requirements. The power budget is based on a 12 Hz carrier loop bandwidth. This means that for the orbital phase some form of rate aiding is required at the ground station or orbital tracking is severely constrained since a 48 Hz bandwidth is required for periods of maximum radial acceleration (reference Appendix B, Metric Analysis).

The telemetry link is capable of operating in a number of modes, The various operating modes include a choice of 1 - or 50-watt transmitter powers and a choice of low, medium, or high gain spacecraft antennas. In addition, at the receiving terminal the system may operate either with the 210-foot Deep Space Network or the 85-foot Deep Space Network antenna. Figure 2-4 is a nomograph showing the performance margin as a function of communication distance for 512 bit/sec telemetry plus pseudo random noise ranging with the 210- foot Deep Space Network antenna. Figure 2-5 is a similar nomograph for 8 bit/sec telemetry only. Figures 2-6 and 2-7 show the equivalent performance with the 85-foot Deep Space Network antennas. A listing of the various spacecraft transmission modes is given in Table 2-8. The table gives the normal usage of each mode and the maximum communication distance assuming the sum of the negative tolerances for each mode.

Table 2-7. Downlink Power Budget for 8 bits/sec Data*

Mode: Spacecraft Low Gain Antenna, 210-Foot DSIF Antenna

Communication Distance: 240 x 10^6 Kilometers (Nominal Encounter)

No.	Parameter	Value	Tolerance +	Tolerance -
1	Transmitter Power	+47.0 dbm	1.0	0.0
2	Transmitting Circuit Loss	-2.1 db	0.8	0.8
3	Transmitting Antenna Gain**	0.0 db	0.3	0.5
4	Transmitting Antenna Pointing Loss	0.0 db	-	-
5	Space Loss 2295 MHz at 240 x 10^6 km	-267.4 db	0.0	0.0
6	Polarization Loss	-0.1 db	-	-
7	Receiving Antenna Gain	61.7 db	1.0	0.5
8	Receiving Antenna Pointing Loss	-0.5 db	-	-
9	Receiving Circuit Loss	-0.2 db	0.1	0.1
10	Total Received Power	-161.6 dbm	3.2	1.9
11	Receiver Noise Spectral Density T System = 45 ± 10°K	-182.0 dbm/Hz	1.1	0.9
	Carrier Performance			
12	Modulation Loss	-2.4 db	0.3	0.3
13	Received Carrier Power	-164.0 dbm		
14	Received Carrier SNR ($2B_L$ = 12 Hz)	7.2 db		
15	Sum of the negative tolerances	3.1 db		
16	Worst case carrier SNR	4.1 db	0.0 db	3.1 db Peak (1.4 db RMS)
	Engineering Telemetry Performance (8 bits/sec)			
17	Modulation Loss (θ = 1 radian)	-3.8 db	0.8	0.8
18	Demodulator Degradation	-1.5 db	-	0.5
19	Received Subcarrier Power	-166.9 dbm		
20	Received SNR (8 bits/sec)	6.1 db		
21	Required SNR (5 x 10^{-3} error rate 32/6 code)	2.0 db		Peak
22	Margin	4.1 db	5.5 db	4.1 db (1.7 db RMS)

*Downlink power budget for bit/sec data illustrates communications constraint imposed by the downlink carrier when transmitting over the low gain antennas. Carrier performance margin indicates that for a conservative design communications can not be maintained substantially beyond 240 x 10^6 kilometers with omnidirectional coverage.

**Based on 90 percent coverage by the combined low gain antennas.

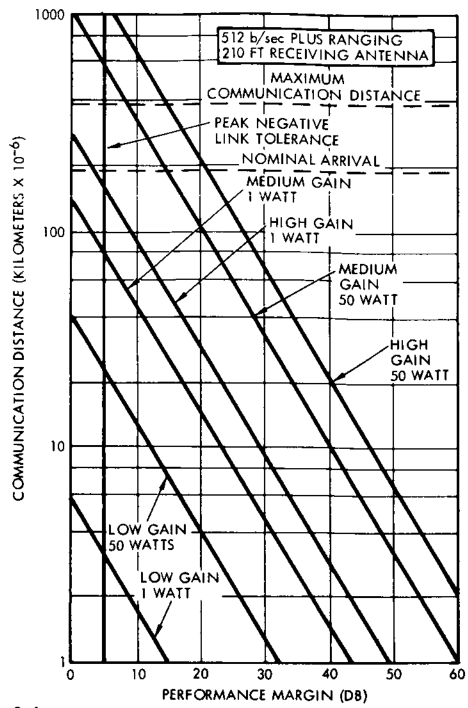


Figure 2-4

NOMOGRAPH OF COMMUNICATION DISTANCE shows the performance margin available as a function of communication distance for various spacecraft transmitter and antenna combinations operating into a 210-foot DSN ground station antenna. The curves are based on simultaneous transmission of 512 b/sec telemetry data and ranging.

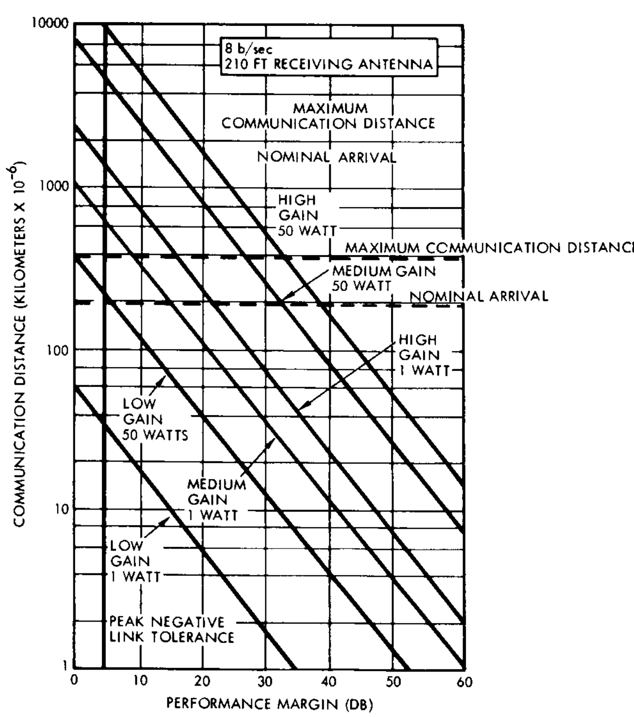


Figure 2-5

NOMOGRAPH OF COMMUNICATION DISTANCE versus performance margin for 8 b/sec telemetry into a 210-foot DSN ground station antenna.

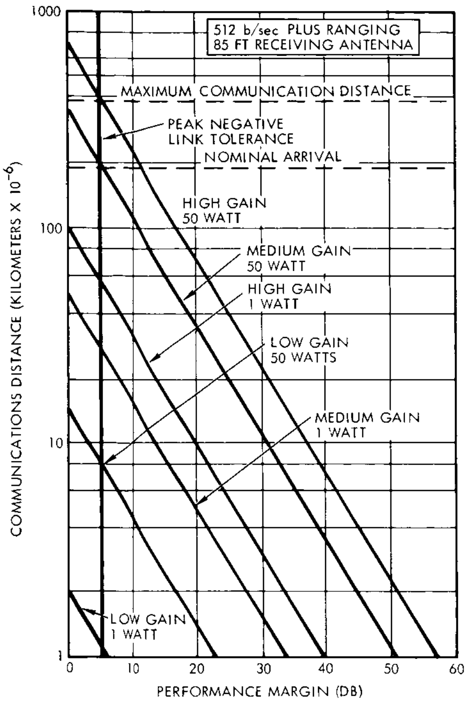


Figure 2-6

NOMOGRAPH OF COMMUNICATION DISTANCE versus performance margin of 512 bit/sec telemetry into an 85-foot DSN ground station antenna.

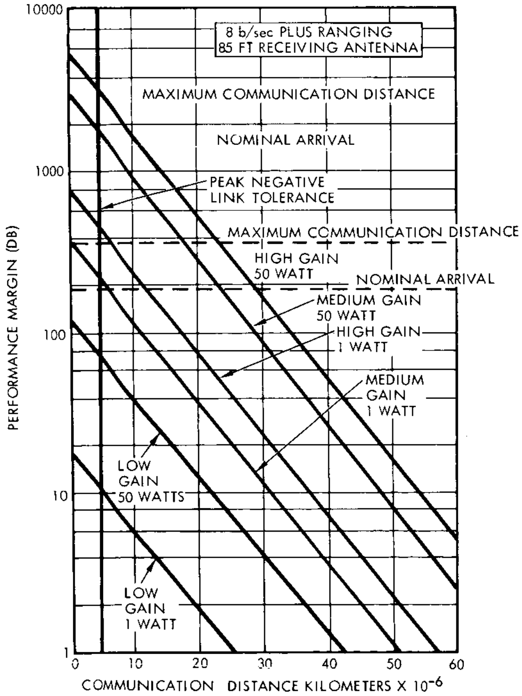


Figure 2-7

NOMOGRAPH OF COMMUNICATION DISTANCE versus performance margin for 8 b/sec telemetry into an 85-foot DSN ground station antenna.

Table 2-8. Various Downlink Transmission Modes*

Antenna	Transmitter (watts)	Normal Use	Maximum Communication Distance (210-foot Antenna) km x 10^6 ** 8 bits/sec	512 bits/sec Plus Ranging	Remarks
Low Gain (either)	1	Prior to initial cruse orientation	25	3.1	
Low Gain (either)	50	Early cruise and during reorientation maneuvers	240	23	Carrier track only beyond 240 x 10^6 km
High Gain	1	Cruise phase	1320	160	Backup to 50 watt transmitter during orbital operation
High Gain	50	Orbital operations	9600	1200	
Medium Gain	1	Backup to high gain and during pre-maneuver phase	660	80	
Medium Gain	50	Backup to high gain during orbital operations	4800	590	

*Various downlink transmission modes are employed depending upon the mission phase. The maximum distance over which diagnostic telemetry data can be transmitted depends on the selected mode and the data transmission rate. During orbital operations, the directional antenna modes are used for telemetry transmission.

**Based on peak negative link tolerances.

For normal operations the spacecraft transmits 512-bit/sec telemetry data and ranging using the 1-watt transmitter over the low gain antenna until spacecraft cruise orientation has been achieved. This made can be supported out to 1 million km with the 85-foot antenna or 3 million km from the 210-foot antenna. After orientation has been achieved, the data is switched to the 50-watt transmitter and the high gain antenna. This mode continues until just before midcourse maneuvers, when transmission is switched to the medium gain antenna so that the high gain antenna can be slewed to be pointing earthward after the orientation maneuver has been completed. Both the high gain and medium gain antennas are capable of supporting 512 bit/sec telemetry and ranging out to encounter with the 50-watt transmitter and the 85-foot ground antenna.

The downlink spectrum from each spacecraft requires 3 MHz bandwidth. Restriction of the bandwidth to 3 MHz is made possible by truncating the pseudo random noise ranging signal at 800 kHz. This technique has been demonstrated on previous programs and does not produce significant degradation of the ranging data. Two 3 MHz downlink channels plus a 1 MHz guard band use up 7 MHz of the available 10 MHz. This leaves 3 MHz for the two capsule data links. While this should be quite satisfactory for a normal mission, it does not provide channel space for a spare spacecraft. Two approaches are being considered to alleviate this problem. One approach is to provide two crystal oscillators in each receiver and transmitter selectable by ground command. This permits either spacecraft or the spare to operate on either channel. The second approach is to place the spare spacecraft channel midway between the two assigned channels and in the event the spare spacecraft was used, the high rate data would have to be time-shared between the two spacecraft. A typical downlink spectrum for two spacecraft and two capsules is shown in Figure 2-8.

2.2.3 Tracking

The S-band radio subsystem retransmits the received signal to permit measurement of spacecraft range and range rate by the Deep Space Network. Range rate is determined from the doppler shift obtained by retransmitting a carrier frequency coherently related to the spacecraft received frequency by a ratio of 240:221. This downlink frequency is then

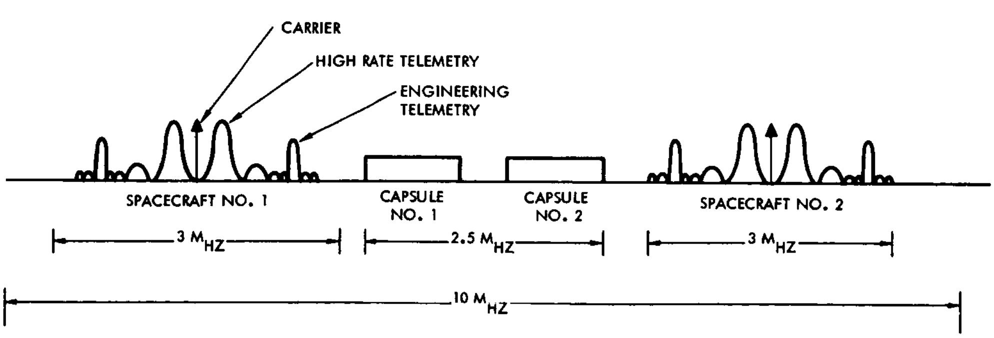


Figure 2-8
DOWNLINK SPECTRUM showing possible arrangement for two spacecraft and two capsules.

compared with the ground transmitted frequency to derive the doppler shift. The ranging measurement is accomplished by measuring the phase delay of a pseudo random noise code sequence made up of five subcode words and has the property of requiring a correlation (acquisition) time proportional to the sum of the bits in the five code words and an unambiguous length proportional to the product of the bit count in each code word. The code has the advantage that it has a large unambiguous range measurement capability while at the same time it is capable of being acquired within a reasonable period of time. It has the disadvantage that it requires a transmission bandwidth of at least 2 MHz.

2.2.3.1 Ranging

The primary impact of the pseudo random noise ranging signal on the S-band radio subsystem is in the receiving and transmitting bandwidths required to accommodate the signal. This affects equipment bandwidths and the spectrum utilization within the allocated 10 MHz Deep Space Network bandwidth.

Essentially, two methods may be employed for receiving and retransmitting the pseudo random noise ranging code. In the first method, the code is demodulated by the receiver and regenerated by an on-board decoder. The second method uses a direct turnaround method in which the demodulated code is processed through a post-detection filter and remodulated directly onto the downlink carrier. The direct turnaround approach has the advantage of simplicity but the disadvantage that noise is turned around with the ranging signal, which essentially robs power from the downlink carrier. Also, predetection suppression of the uplink signal results in an effective reduction of the downlink modulation index as a function of uplink received signal strength. The regeneration technique does not have the disadvantages of turnaround noise and suppression, but it does have the disadvantage that it requires fairly extensive on-board processing equipment and also requires a substantial amount of time to acquire the uplink code prior to retransmission.

For the Voyager application, the two systems have been compared (see Appendix B) and the direct turnaround method has been selected.

The selection has been based on the fact that the available downlink ERP limits the ultimate distance over which the range measurement can be achieved. In order to make a range measurement at the maximum range of 400×10^6 km, it is necessary to employ the high or medium gain antenna for downlink transmission. Since this implies that the antenna must be properly pointed, it is possible to use the same antennas for uplink reception. Under this condition, direct turnaround ranging can be supported. This requires that receivers be provided for the directional antenna channel. However, they are less complex than the corresponding code-regeneration equipment and they have the added advantage of providing redundant paths for command and doppler reception. The comparison between turnaround ranging and regenerated ranging is illustrated in Table 2-9, which indicates the maximum communication distance for each method using the low gain antenna. It also gives the performance margins available at the end of the mission for the medium and high gain antennas. It is apparent that with the low gain antenna, the ranging measurement can not be achieved with either approach and that with the directional antennas the margins are more than adequate for the entire mission.

Table 2-9. Comparison of Turnaround Versus Vehicle Regeneration of Range Code*

	Maximum Range with Omni and 6 db Margin (km)	Margin at Encounter + 6 m	
		Med Gain (db)	High Gain (db)
Turnaround Ranging	11.5×10^6	10.5	16.1
Vehicle Regeneration of Range Code	15×10^6	10.7	16.7

*Comparison of Turnaround versus Vehicle Regeneration of Range Code shows that very little advantage is obtained by reconstructing the range code in the spacecraft considering the hardware complexity involved.

2.2.3.2 Range Rate

The doppler tracking requirement essentially dictates the operational characteristics of the receiver phase lock loop, (see also Support Analysis in Appendix B.) The minimum phase lock bandwidth is inversely proportional to the square of the radial acceleration. The Voyager mission differs from earlier deep space mission in that a significant radial acceleration exists during the orbital phase when the spacecraft velocity vector is normal to the earth-Mars line of sight. The maximum acceleration is of the order of 2.5 meters/sec^2. With this acceleration the threshold phase lock bandwidth has been calculated at 32 Hz for the spacecraft and 48 Hz for the ground receiver. The vehicle receiver bandwidth does not pose any significant problem in terms of implementation or receiver sensitivity requirements. However, the downlink bandwidth means that some form of rate adding may be required in the ground receiver when downlink transmission over the low gain antenna is used in orbit.

2.2.4 Redundancy and Failure Modes

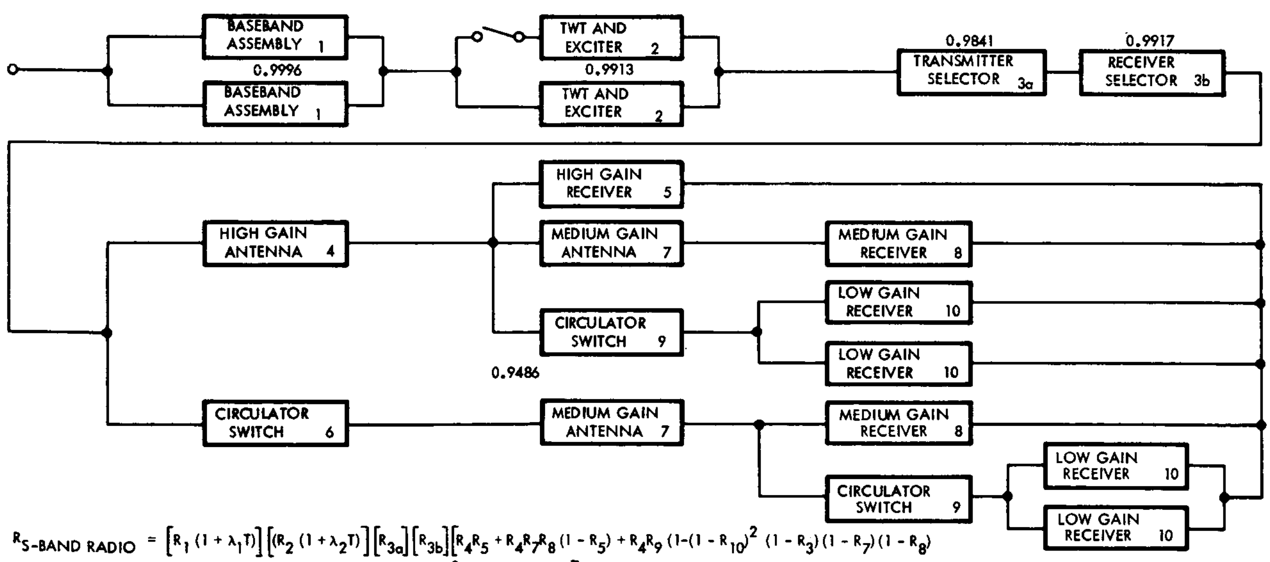
Redundancy is accomplished by the use of either redundant modules or redundant circuitry. The redundant components are connected either by switches or by cross-strapping the inputs and outputs. Replacement of failed components is achieved either by ground command or by automatic failure sensing circuitry. The type of redundancy for the various equipment is summarized below.

2.2.5 Reliability Assessment

The reliability estimate for the S-band radio subsystem is 0.916. Figure 2-9 presents the reliability equation and block diagram of the subsystem. Detailed calculations are presented in Vol 2 Appendix E Section 3.8.6.

Equipment Redundancy

	Redundant Mode	Connection Method	Remarks
Antenna	Module	Switching	
Baseband Assembly	Circuitry	Cross-Strapping	
Exciter Modulator Power Amplifier	Module	Cross-Strapping the inputs and switching the outputs	
Power Diplexer and Circulator Switch Assembly	None	None	Passive elements only, inherently reliable
Preamplifier - Phase Lock Receiver	Module	Switching	Four are used, one for each antenna
Receiver selector and low gain antenna selector	Circuitry	Cross-Strapping	



$$R_{S\text{-BAND RADIO}} = [R_1 (1 + \lambda_1 T)] [(R_2 (1 + \lambda_2 T)] [R_{3a}] [R_{3b}] [R_4 R_5 + R_4 R_7 R_8 (1 - R_5) + R_4 R_9 (1-(1 - R_{10})^2 (1 - R_3)(1 - R_7)(1 - R_8)$$
$$+ R_6 R_7 R_8 (1 - R_4) + R_6 R_7 R_9 (1 - (1 - R_{10})^2 (1 - R_8)(1 - R_4)] = 0.916$$

Figure 2-9

THE RELIABILITY BLOCK DIAGRAM OF THE S-BAND SUBSYSTEM shows the redundancy of critical elements and the possible alternate modes of operation in case of failure of certain units.

2.3 COMPONENT DESCRIPTION

2.3.1 High Gain Antenna

The high gain antenna (Figure 2-10) is a focus fed circular paraboloid reflector with 2 degrees of pointing freedom. It has 34 db nominal gain at 2295 MHz. The reflector has a 9.5 foot diameter aperture, illuminated by a short backfire primary radiator at the parabola focus. The primary radiator generates left-hand circular polarization that is reflected into right-hand circular polarization. To obtain low axial ratio off-axis, fins are added to the outer ring of the primary radiator (Figure 2-11). In this way, the electric and magnetic plane patterns of each dipole are equalized. Tests

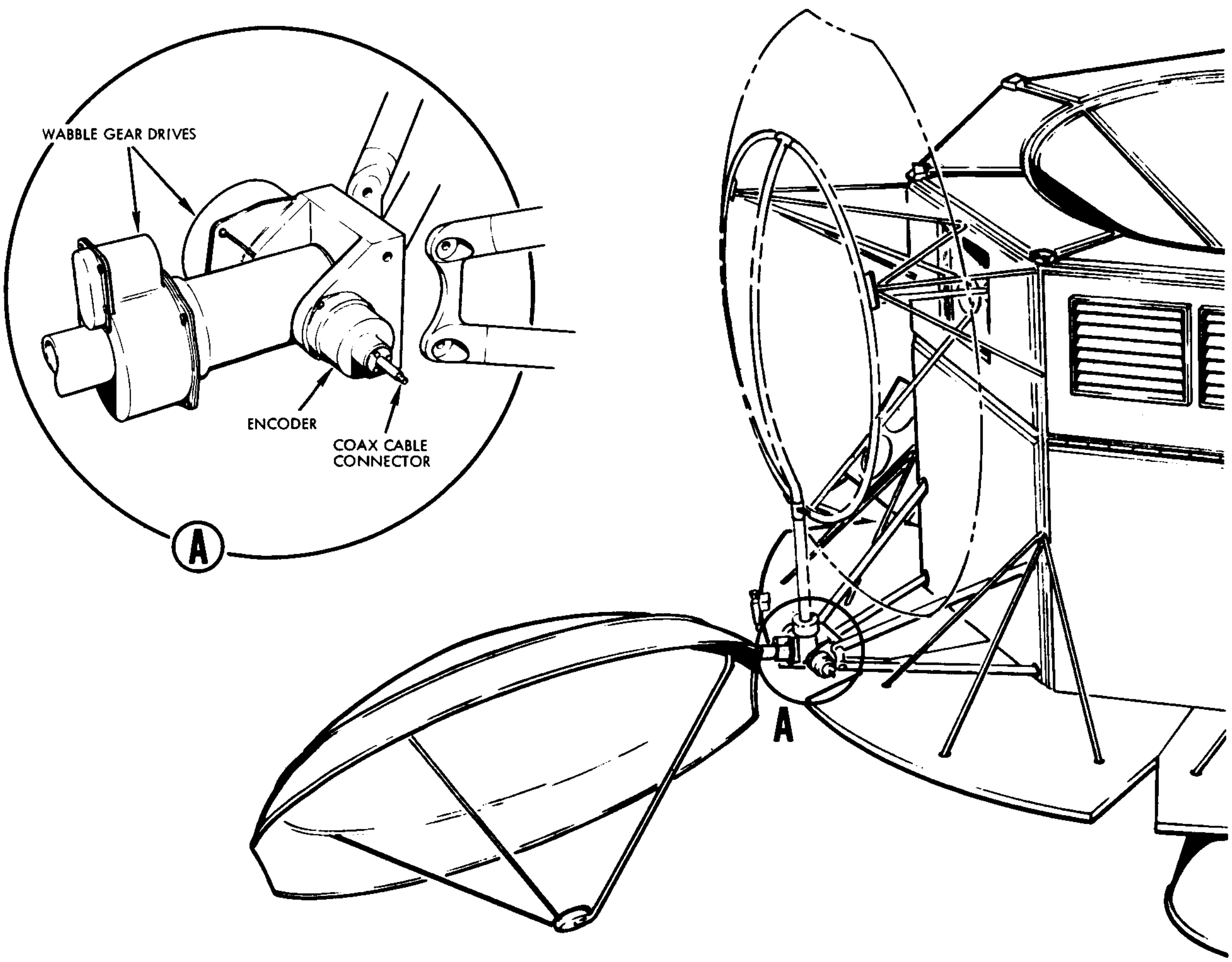


Figure 2-10
THE HIGH GAIN ANTENNA OF THE RECOMMENDED SPACECRAFT stows on the side of the equipment module. It has two wabble gear drives that enable it to be adjusted to any desired pointing angle.

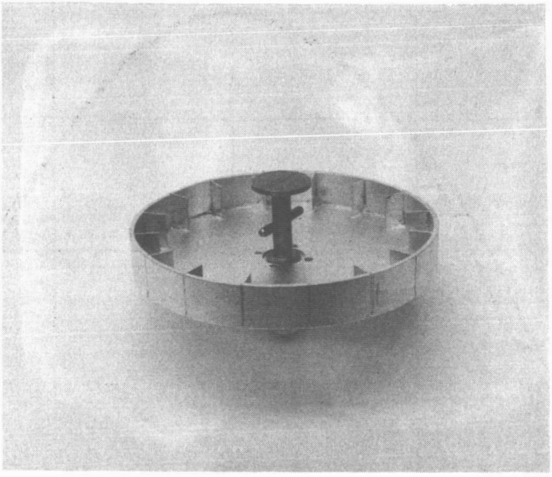


Figure 2-11

EXPERIMENTAL SHORT BACKFIRE PRIMARY RADIATOR for high-gain antenna used to demonstrate pattern equalization by fins around periphery.

with an experimental model show good equalization down to - 10 db on each side of the main beam (Figure 2-12).

Maximum gain is obtained from the antenna by using a 10 db edge taper in the illumination of the parabola by the primary radiator. The resulting beamwidth is 3.2 degrees with a 0.43 focal length to diameter ratio.

The reflector is made from 3/4 inch honeycomb sandwich construction. The feed is held by a tripod of aluminum alloy tubes. This tripod is also part of the tie-down in the stowed position. A 70-inch diameter ring, of 2-inch aluminum alloy tubing joins the antenna to the support arm that connects to the drive mechanism.

In the stowed position the antenna feed tripod fits into a recess in the upper portion of the equipment module. The meteroid panel serves as a structure for the mounting of the release device, which is tension loaded to the feed structure. Two bearing pads, one on either side of the reflector, support the antenna in its stowed position.

Table 2-10 shows the results of a tradeoff study from which the focus feed parabola was selected—for maximum gain per pound and lowest development risk involved.

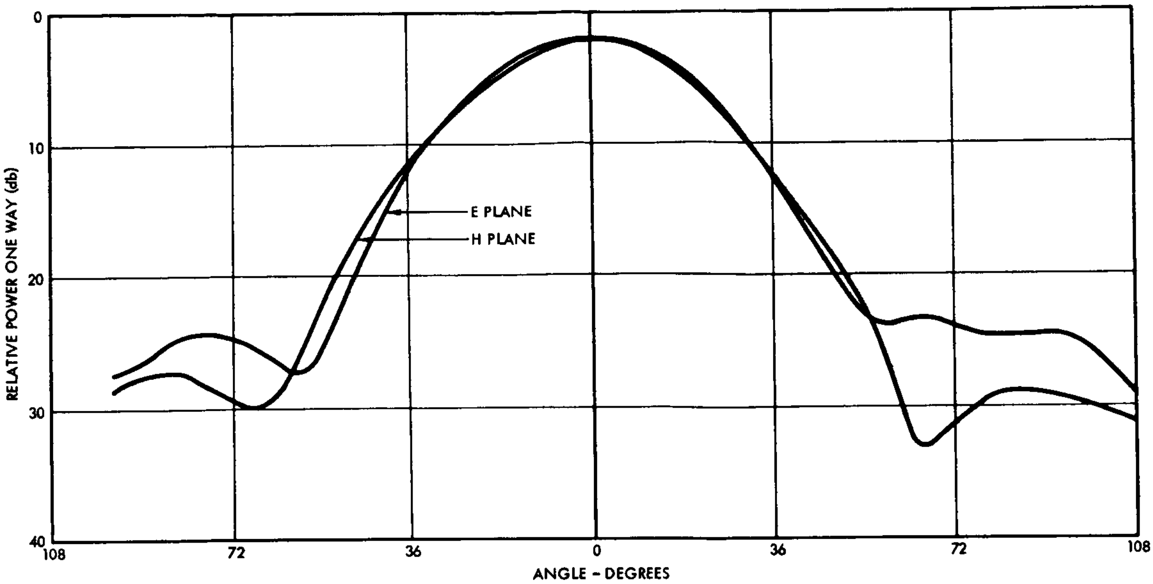


Figure 2-12
MEASURED PATTERNS OF EXPERIMENTAL PRIMARY RADIATOR show good equalization between E and H planes.

In the gain region between about 25 db and 40 db the focus fed parabolic reflector usually is the best choice for any application which does not require rapid scanning or widely separated multiple beams. Below 25 db the phased array shows advantages in higher aperture efficiency without excessive loss in the feed network.

Using a phased array in an application requiring an 8 percent bandwidth requires that the feed network be arranged so that every element is fed by an equal length of transmission line. This requires a series of power dividers each with a loss of about 0.1 db. After 6 or 7 levels of power dividers the losses are comparable to that of the parabola. Using two way dividers, this is equivalent to 64 or 128 elements (2^6 or 2^7). To avoid grating lobes the elements must be less than a wavelength apart. Assuming 3/4 wave spacing, the resulting square aperature would be less than about 8 wavelengths on a side. A uniformly illuminated aperture of this size results in a 6 1/2 degree beam width and a gain of about 28 db.

The cassegrain reflecting system begins to be attractive for gains above 40 db. The main limitation of the cassegrain for lower gains (near 40 db) is the subreflector size. For geometrical optics to apply, the subreflector must be at least seven wavelengths in diameter, but it must be

Table 2-10. Tradeoff Study of High Gain Antennas Indicates Parabola to be Best Choice

Antenna Type	Typical Efficiency (%)	Typical Band-width (%)	Gain 9.5 ft Aperature	Weight (est)	Gain (db/lb)	Advantages	Disadvantages
Parabole-focus fed	60	10	33.7 db	28.9	1.165	Highest gain/lb lowest development risk.	Unmodified shape is difficult to stow
Phased array of dipoles - equi-length stripline feed network two way power dividers	67	10	34.2	47	0.728	Mod. gain-low volume	Requires additional materials research to lighten network design
Phased array of slats waveguide feed - standing wave mode	70	2	34.4	55	0.625	High gain	Narrow bandwidth
Phased array of slots traveling wave mode Radial trans. line	65	8	34.05	61	0.56		Extra weight and loss from dissipa-tive load
Cassegrain reflector	65	10	34.05	32	1.06	Shorter cable run to feed	Aperture too small to get indicated efficiency performance - Required positioning extra reflector

at least seven wavelengths in diameter, but it must be small relative to the diameter of the main reflector. If the loss from blockage can be as high as 5 percent, the main reflector must be at least 30 wavelengths in diameter with a gain near 40 db. Cassegrain antennas have been described with subreflectors as small as 4 wavelengths using a dielectric guide between the primary radiator and the subreflector. Difficulties with the dielectric material has prevented good performance over an 8 percent bandwidth with low axial ratios. The development risk involved with this approach is considered to be sufficient to favor the standard parabola for the Voyager application.

2.3.1.1 High Gain Antenna Drive

The high gain antenna drive mechanism rotates the antenna about the shaft axis and the hinge axis. Its preliminary specification is given in Figure 2-13. The antenna is pointed to earth by using both drives. The hinge axes drive is also used to deploy the antenna. Antenna pointing direction is updated by ground commands or by programmed commands from the computer and sequencer. These commands enter the antenna drive electronics and control the power feed to the two-phase AC servomotor used as the prime mover. The motor output is geared down

PRELIMINARY SPECIFICATION

High Gain Antenna Drive

Performance Characteristics

GIMBAL TRAVEL	
Hinge axis:	± 90° plus stowage
Shaft axis:	360° unrestricted
GIMBAL RATES	
Slew:	5.3 milliradians per second (minimum each axis)
Angular acceleration:	0.5 milliradians per second^2 (minimum each axis)
LOAD TORQUES *	
Hinge axis:	3350 in./lb
Shaft axis:	216 in./lb

* The double gimbal drive mechanism is designed for the deployment and shaft axes of the Planetary Scan Platform, which will carry slightly higher loads than the antenna drive axes.

INPUT VOLTAGES	
Drives:	36 volt
Heaters	28 VDC
POWER	
Hinge axis:	51 watts running, 73 watts peak
Shaft axis:	9.1 watts running, 11.4 watts peak
Pointing accuracy:	± 4 milliradians relative to the spacecraft
Encoder:	± 0.10° (three encoders will be provided for each axis)
RF path:	two noncontacting rotary joints
Drive weights:	5.3 lb (hinge axis), 3.8 lb (shaft axis)
Seals:	high speed elements hermetically sealed in inert atmosphere
Magnetic requirement:	1000 gamma at one foot of an equivalent dipole, operating and nonoperating
Antenna weight:	30 lb

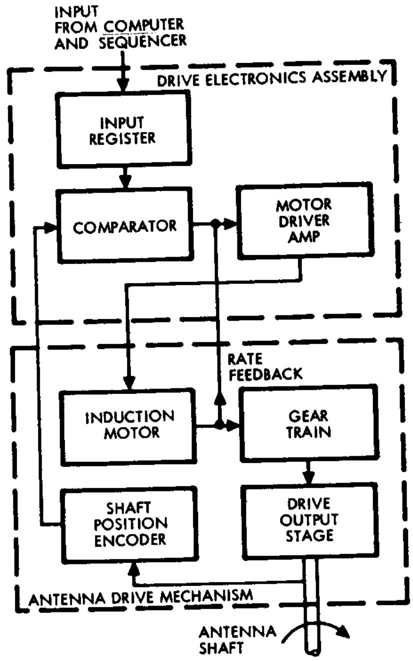


Figure 2-13

through a sealed drive whose final mechanism is a wabble gear that rotates the antenna shaft. Finally, a digital encoder sends a position feedback signal to the electronics in the computer and sequencer subsystem. Three encoders are provided for redundancy. A size 15 servo motor drives the hinge axis, and a size 23 servo motor, the shaft axis.

The design (Figure 2-14) chosen for the high gain antenna drive assemblies consists of a cross gimbal housing, an RF joint and two wabble gear drive mechanisms. The wabble gear drive was selected because of extensive space experience and background accumulated in it over the past five years. Its success in long-life space missions and its relative simplicity are strong advantages. However, the harmonic drive will be retained as a design alternative.

The assembly makes a neat, compact package which can be fabricated, tested, and handled as a unitized spacecraft component. The 36-volt two-phase drive motor and wabble gear are housed together to form a sealed drive. There are no dynamic seals in this design. All

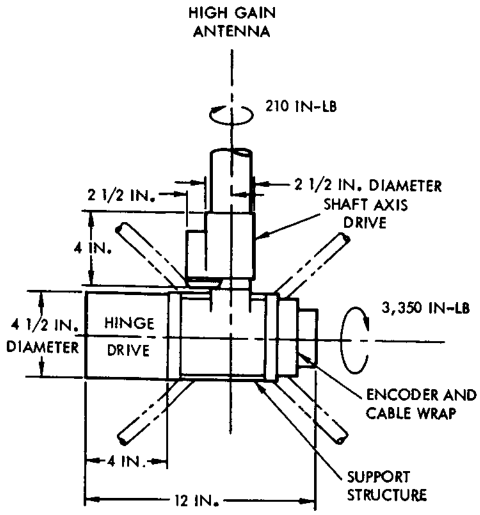


Figure 2-14
THE HIGH GAIN ANTENNA DRIVE includes a shaft-axis drive and a hinge drive. Total weight of drives is 5.5 pounds.

rolling and sliding surfaces of the motor, gearhead, associated gears and ball bearings are hermetically sealed in pressurized inert gass for protection against a vacuum environment. The exceptions are one pair of gears and one ball bearing pair, specially designed to operate at low speed in vacuum. The lubricants used in the sealed drive are radiation resistant. Sealing is accomplished by two bellows installed between the nonmoving parts of the mechanism and the driving gear.

The servo motor is a high-resistance, induction motor, often referred to as a 2 phase control motor. It consists of a stator with two windings displaced 90 electrical degrees from each other, and a squirrel-cage rotor (or equivalent). Each winding is excited by its own voltage source; these voltages differ in phase by 90 degrees. The first phase is called the fixed phase and the voltage is of constant amplitude and constant frequency. The other phase is called the control phase and the voltage is of the same frequency but varies in amplitude.

2.3.1.2 Power Requirements

The load on the high gain antenna shaft axis drive is made up of the following components:

Inertial load:	7.26 in/oz (assuming antenna to be a flat disc of uniform mass)
Bearing friction:	100 in/oz (estimated)
Mass unbalance:	3360 in/oz
Total:	3467 in/oz

With a gear ratio of 6000:1 and an estimated gearing efficiency of 50 percent, the required motor torque is 1.15 in/oz. Using a size 15 AC servo motor, this results in a peak power requirement of 11.4 watts and running power requirement of 9.1 watts. The control winding amplifier is required to deliver up to 5.3 watts of power.

The load on the hinge axis drive consists of the following:

Inertial load:	36.4 in oz
Bearing friction:	100 in oz
Cable wrapup:	200 in oz (10 wires)
Mass unbalance:	53,700 in oz
Total:	54,000 in oz

The design calls for a gear ratio of 30,000:1 and a size 23 AC servomotor for the hinge axis; this results in a required motor torque of 3.6 in oz. The resulting peak and running power requirements are 73 and 51 watts respectively. The amplifier used to provide power to the servomotor control winding must be capable of supplying up to 33 watts. The cable wrapup must be heated to at least $10^{\circ}F$ during operation and will require approximately 6 watts of additional power.

The unique feature of the driving gear mechanism is the use of a pair of specially cut wabble gears for the output stage (Figure 2-15). Action of the bearing carrier and a tilted bearing internal to the unit produces a nonrotating conical nutation motion of the driving gear at the end of the main bellows. This motion causes rotation of the output gear and shaft by sequential engagement of the gear teeth.

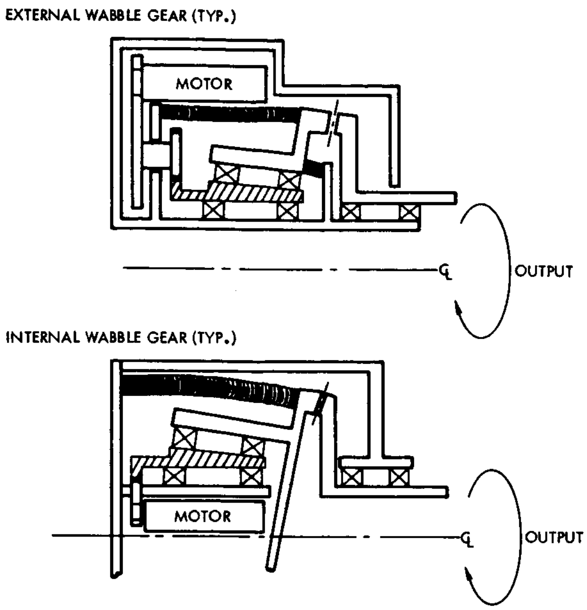


Figure 2-15
TYPICAL WABBLE GEAR DRIVES. The action of the bearing carrier and a tilted bearing internal to the unit produces a non-rotating conical nutation motion of the driving gear at the end of the main bellows. This motion causes rotation of the output gear and shaft by sequential engagement of the gear tooth.

The RF rotary joint is a noncontacting, low loss transmission line joint. The antenna shaft/mount, RF joint, and wabble gear drive are all concentric. In the high gain antenna, the hinge axis drive provides for a cable wrapup to carry power and signals to the shaft axis drive. Heaters are located in the wabble gear housing to maintain drive and cable wrapup temperatures above $10^{o}F$ during operation.

The position pickoff used for control system feedback is an incremental encoder. An RF rotary joint in each axis of rotation transmits energy between the high gain antenna and the equipment compartment. To keep insertion loss as small as possible and give rotation at the extreme temperatures of space environment, use is made of air dielectric, noncontacting, choke-coupled rotary joints. Their insertion loss is less than 0. 10 db over the operating frequency band of 2000 to 2500 Mc. Mechanically, the rotary joint will be an integral part of the gimbal axis structure, relying on the gimbal axis structure, relying on the gimbal structure for alignment as well as the rotating mechanism.

The electrical operation of the rotary joint can be seen schematically in Figure 2-16.

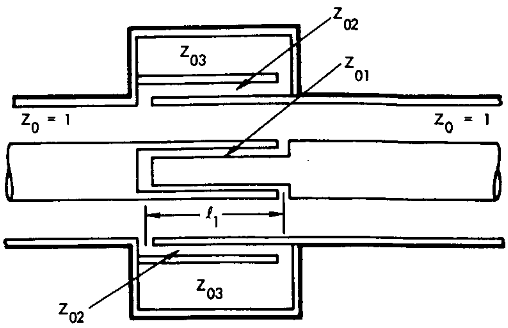


Figure 2-16
THE ROTARY JOINT is shown schematically indicating the electrical operation. The choke sections are labeled 1. Z_{03} is an external choke impedance.

All the choke sections, denoted by 1, are one quarter-wave long at the center frequency. To prevent leakage, an external choke of impedance Z_{03} is added to the outer conductor choke section.

The antenna drive electronic circuitry is shown in Figure 2-17. The desired antenna angle relative to spacecraft coordinates received

PRELIMINARY SPECIFICATION

High Gain Antenna Drive Electronics

Purpose

Processes inputs from the computer and sequencer, and provides drive power to the high gain antenna drive actuator to about two axis. Antenna position readout is provided to the telemetry system.

GENERAL INFORMATION

Digital protion is triple redundant on a subsystem level.

HIGH GAIN ANTENNA DRIVE ELECTRONICS INPUTS

C and S commands (two axis)

HIGH GAIN ANTENNA DRIVE ELECTRONICS OUTPUTS

Two position-drive signals to hinge and shaft axis actuators

Telemetry position pickoff of hinge and shaft axis actuators

Physical Characteristics

Power:		50 VDC
Hinge-Axis:	Peak	8 watt
	Minimum	7 watt
Shaft-Axis:	Peak	5 watt
	Minimum	5 watt
Size:		7 x 6 x 4 in.
Weight:		5 lb

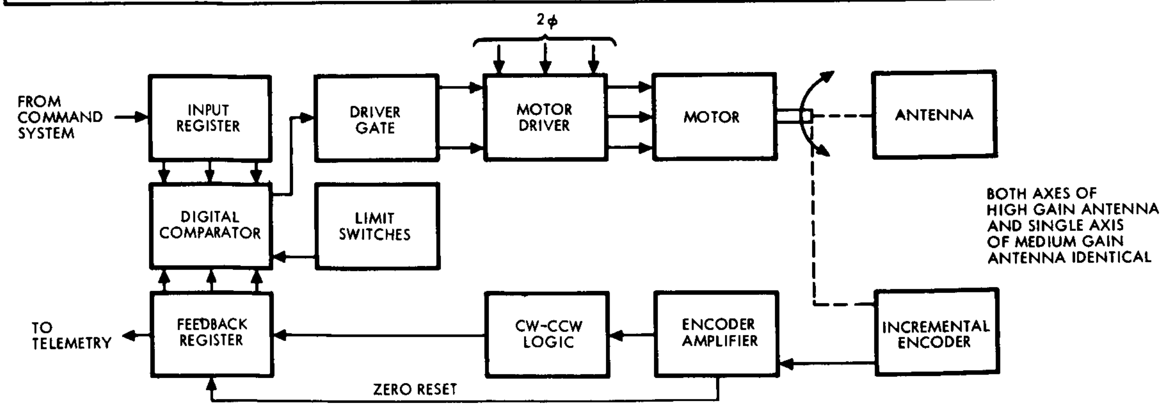


Figure 2-17

from the computer and sequencer enters an input register. This data is compared with that in the feedback position register in the digital comparator. If the two registers do not compare in every bit, the comparator delivers a signal of proper polarity into the driver gate. This turns on the motor drivers, which apply full power to the antenna drive motor. The polarity of the driver-gate input dictates the direction the motor rotater.

Antenna position information comes from the incremental encoder mounted to the antenna shaft. Output pulses from the encoder are entered into the feedback register, which will constantly accumulate the absolute antenna angle. This information is then used for comparison into the digital comparator and for telemetry.

A zero reset capability is provided by the encoder. When the encoder passes through null, a pulse is emitted which is amplified and fed into the reset input of the register to assure a zero feedback indication.

The contribution to antenna pointing error by the electronics drive will be less than 0.1 degree. Since the least significant bit of the register used will correspond to 0.1 degree error, this bit will cause the antenna to be driven until the two registers are identical, at which time power to the motor is removed. The total and component high gain antenna pointing errors after in-flight boresight calibration are shown in Table 2-11. The total error is 0.905 degree which is within the 0.95 degree radial error allowance for 1 db loss with a 9.5 foot diameter antenna.

2.3.2 Medium Gain Antenna

The medium gain antenna provides a system gain within 6 db of the high gain antenna, and is simpler to stow and deploy. It has a fan-shaped beam and a single-axis drive.

The medium gain antenna is a single-gimballed horn-fed paraboloid reflector with an oblong aperture. Figure 2-18 shows a perspective view of the antenna assembly. The oblong reflector provides a 3.6 by 9.5 degree beam and a gain of 29 db on the S-band. It was selected after a tradeoff study was made between several types of antenna. Tables 2-12 and 2-13 summarize this study. The main advantages of the antenna selected are

Table 2-11. High Gain Antenna Pointing Error After In-Flight Calibration

Optical Sensor Accuracy	0.1 deg
Antenna Drive Accuracy	0.1 deg
Root-Sum-Squared Error	0.14 deg

Total error per axis with ± .5 deg limit cycle = 0.64 deg

Total error with both axis considered = 0.905 deg

high gain per pound and low development risk. The second choice is the planar array of turnstiles. Some development risk is involved but it might out-perform the parabola if its development were successful.

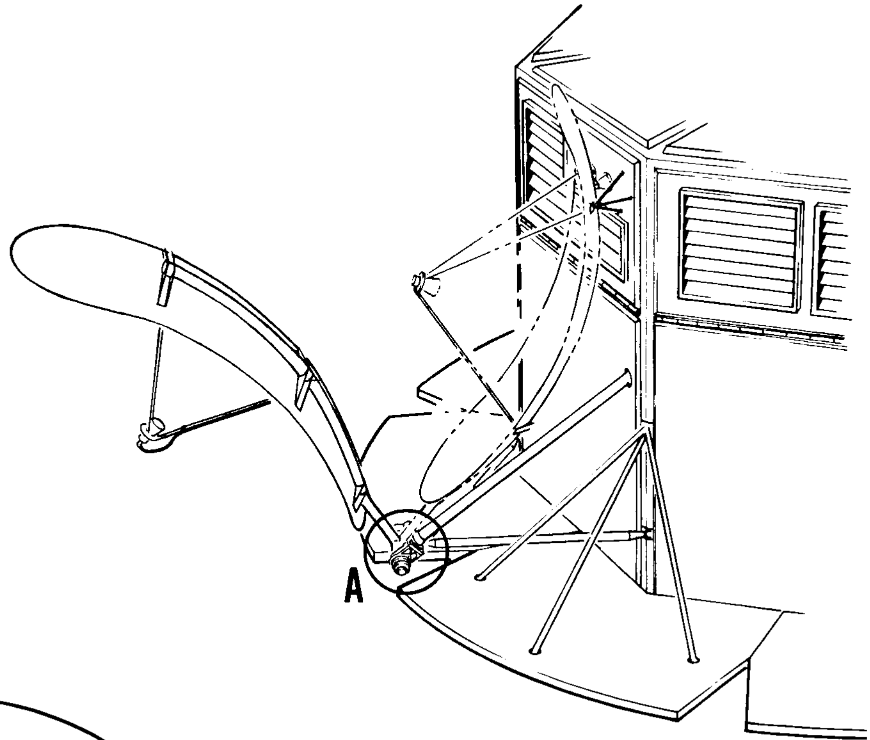
The reflector for the medium gain antenna has a focal-length-to-diameter ration of 0.327 and has been trimmed to a 94 x 36 inch rectangular shape with full end radii. The reflector is 1/2 inch honeycomb sandwich construction consisting of two 0.005 inch aluminum alloy face sheets bonded to a 3/8 x 0.0007 inch hexagonal aluminum core. The feed is supported by a tripod made of one inch aluminum tubes.

In the stowed position the outboard end of the reflector arm is restrained by a tension-loaded release device that preloads the reflector against compression pads.

One rotary joint is required for the single axis gimbal. The same type described in the previous section is used and is mounted within the drive mechanism.

2.3.2.1 Medium Gain Antenna Drive

The medium gain antenna drive is similar to the high gain antenna drive except that it only drives about the hinge axis. It deploys the antenna from its stowed position and points it toward earth. The antenna pointing direction is updated by ground commands or by programmed commands from the computer and sequencer. A size 15, two-phase AC servomotor is used as the prime mover. The output torque of the motor is geared down through a hermetically sealed drive whose final mechanism



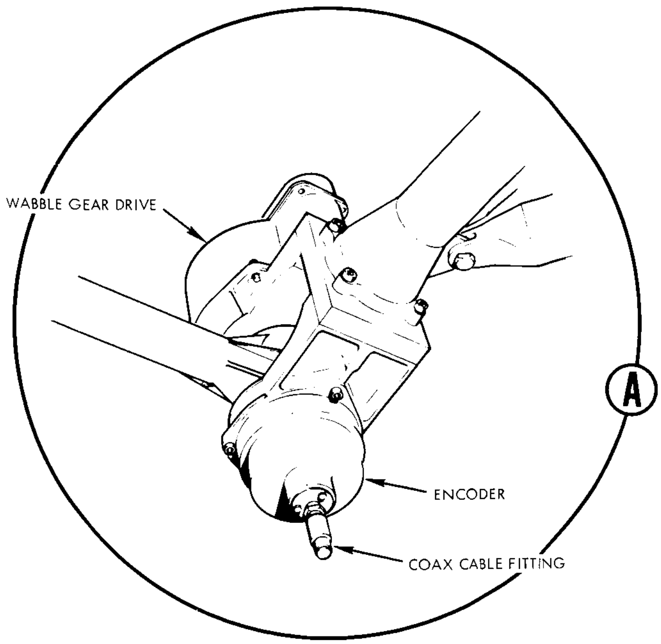


Figure 2-18

THE MEDIUM GAIN ANTENNA is steerable in one axis by means of a single wabble gear drive. It provides a backup to the high gain antenna for the transmission of video mapping data, medium rate science data, and ranging codes.

Table 2-12. Medium Gain Antenna Tradeoff*

Antenna Type	Typical Efficiency (%)	Typical Bandwidth (%)	Gain 36 x 94 inch Aperture	Weight (lb) (radiator only)	Gain/lb	Advantages (for Voyager System)	Disadvantages
Parabola focus feed	55	>10	800	7.7	104	Conservative design highest gain/weight ratio	Unmodified shape is difficult to stow
Phased array stripline fed spirals	60	10	880	11.5	76.5	Small depth, low volume	Weight about 1.5 parabola some development risk
Waveguide fed slots	75	2	1100	12.6	87.4	Small depth, low volume, highest gain	Narrow bandwidth weight about 1.6 times parabola-some development risk
Traveling wave w/g fed slots	60	10	880	15.8	55.7	Small depth	Weight about 2.0 times parabola, development risk
Planar array 9 x 24 turnstiles	70	10	1030	10.5	98	Small depth, good gain/weight ratio	Some development risk
Horn	85	>10				Highest efficiency	Greatest depth, volume, and weight

*Medium gain antenna tradeoff indicates focal point fed parabola is best choice.

Table 2-13. Different Feed Types

Primary Radiator	Efficiency (estimated)	Axial Ratio Variation (estimated db)	Developmental Risk	Remarks
Oblong conical horn	80	With E and H plane equalization <3	Moderate	Requires fins to equalize E and H plane patterns, excessive length
1 x 3 array of turnstiles	85	>5	High	No control of E and H plane equalization for improving axial ratio
1 x 3 array of conical helices	75	<3	Moderate	20 percent bandwidth
Oblong short backfire	80	<3	Moderate	10 percent bandwidth requires fins for equalization
Spiral array	70	<3	Medium	Wide bandwidth

is a wabble gear. The wabble gear rotates the antenna about its deployment hinge. Position feedback signals are provided to the electronics by three redundant incremental shaft encoders. The drive mechanism is shown in Figure 2-19. The drive electronics are essentially the same as for the high gain antenna described in Section 2. 3. 1. 1.

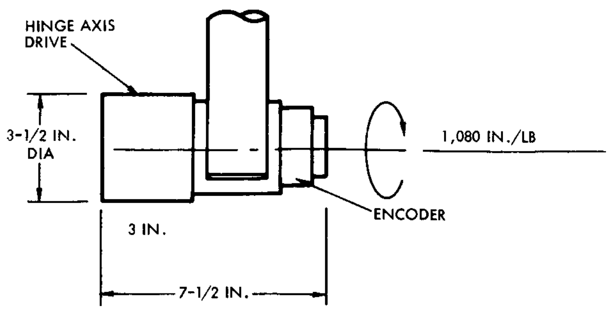
Power Requirements. The load on the medium gain antenna drive is made up of the following:

Inertial load:	0. 832 in oz (negligible)
Friction:	100 in oz in the bearing (estimated)
Mass unbalance load:	17, 300 in oz
Total	17, 400 in oz

With a gear ratio of 40, 000:1 and a gearing efficiency of 50 percent, the required torque is 0. 94 in oz. Using a size 15 AC servomotor, this results in a peak power requirement of 10. 9 watts and a running power requirement of 9. 1 watts. The control winding amplifier is required to handle a peak power demand of 4. 8 watts. The heater for the drive requires 5 watts.

PRELIMINARY SPECIFICATION

Medium Gain Antenna Drive



Performance Characteristics

Pointing accuracy:	± 4 milliradians relative to the spacecraft
Gimbal travel:	$+90^{\circ}$ plus storage
Power:	9.1 watts running, 10.9 watts peak
Gimbal slew rate:	5.3 milliradian/sec
Angular acceleration:	0.5 milliradians/sec^2
Drive weight:	4 lb
Antenna weight:	13.4 lb
Magnetic requirements:	500 gamma at one foot of an equivalent dipole, operating and nonoperating
RF path:	Noncontacting rotary joint

Figure 2-19

2.3.3 Low Gain Antenna

The low gain antenna system provides omnidirectional coverage for each of the S-band radio systems aboard the spacecraft, by use of spatial direction diversity in the vehicle system. The block diagram of the system is shown in Figure 2-20. Two antennas with hemisperical coverage are mounted opposite one another so that the complete sphere around the vehicle is covered by the useful portion of their patterns (Figure 2-21).

A receiver is connected to each antenna. Control logic following the receivers combines the two receiver outputs provided they are above a preselected threshold. If either receiver falls below this value it is inhibited to prevent the possibility of false commands due to a high error rate. This approach provides a smooth transition between the two antennas without degrading the data.

The radiating element (Figure 2-22) consists of four separate conical spiral conductors supported by a fiberglass cone. The spirals are driven with equal-amplitude, phase-quadrature voltages from a printed-circuit hybrid ring located within the cone immediately above the ground

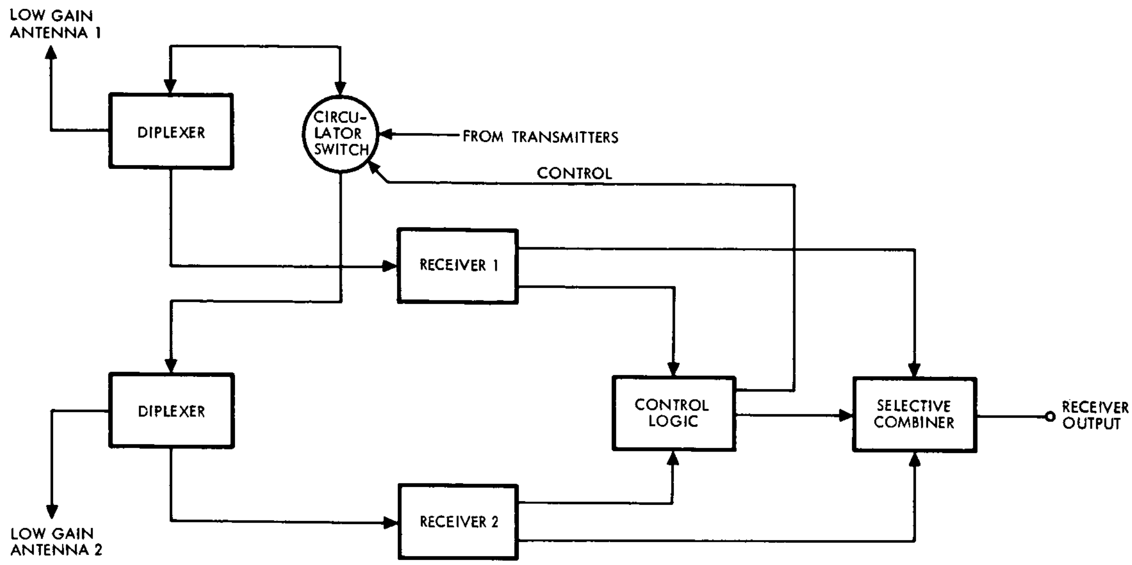


Figure 2-20
LOW GAIN ANTENNA SYSTEM with diversity reception for omnidirectional coverage and selected hemispherical coverage for transmission.

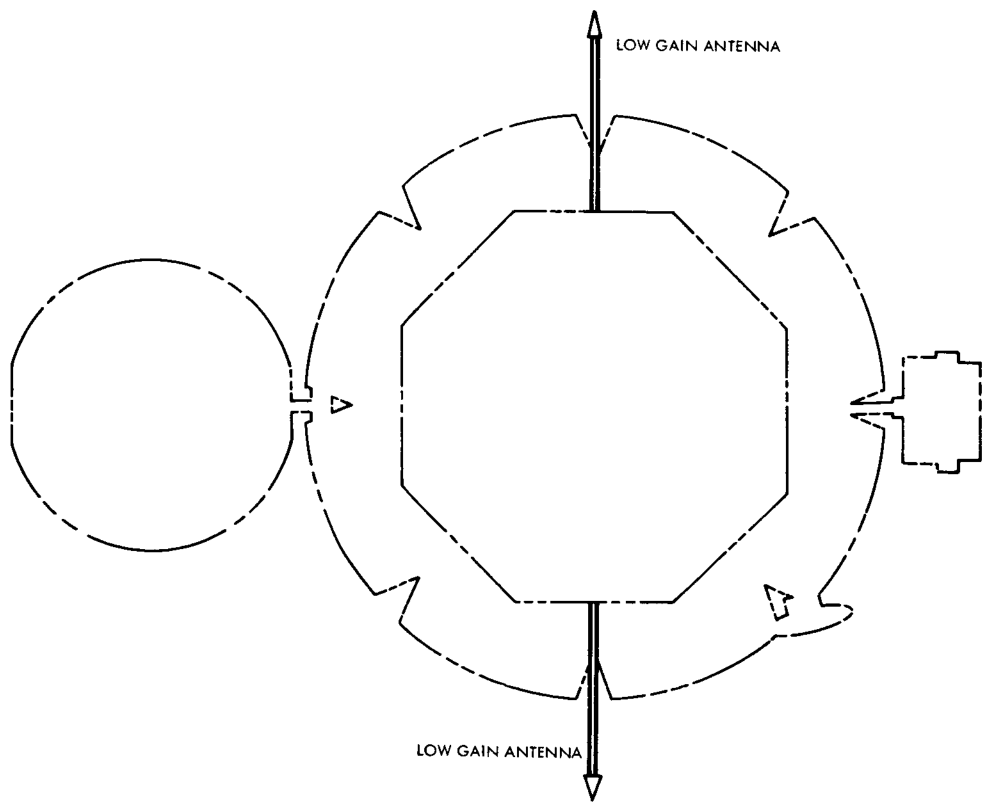


Figure 2-21
TWO LOW GAIN ANTENNAS ON THE SPACECRAFT provide full spherical coverage.

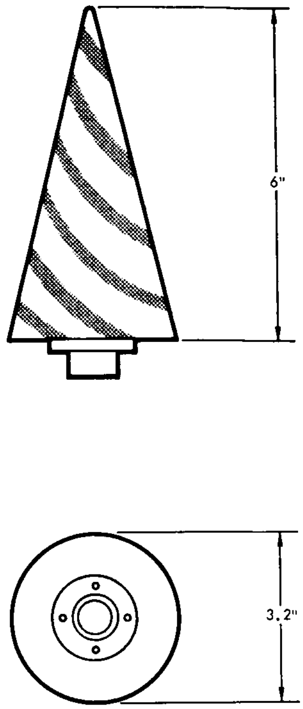


Figure 2-22
LOW GAIN ANTENNA showing a left hand conical spiral to generate right hand circular polarization.

plane base of the cone. The lower side of the ground plane is flange mounted on the end of the support boom.

The antenna produces a right hand circularly polarized wave with an axial ratio less than 2.5 db over a hemisphere. Absolute gain in any direction within the hemisphere is between +1 and -2 db relative to isotropic. If the ground station antenna has an axial ratio of 2 db, an additional polarization loss of up to 0.26 db could occur when the polarization ellipses are not aligned.

This antenna is inherently a broadband device and therefore is not dimensionally critical. Good pattern coverage and impedance match over the 8 percent bandwidth from 2115 to 2295 MHz is well within its capability. Worst-case gain is -2.0 db relative to isotropic and the voltage standing wave ratio (VSWR) is lower than 1.4 to 1. Output impedance is a nominal 50 ohms with coaxial terminals.

Each antenna is mounted on a six-foot tubular boom with a single spring-loaded hinge for deployment. An explosive release mechanism is used to hold the boom in its folded position.

Before deployment the antenna will be used for telemetry and ranging by placing an RF window in the shroud to permit some of the signal to radiate outward from the vehicle. One antenna is secured in a position close to the RF window to obtain the best possible coverage around the vehicle with the shroud in place.

Other diversity systems were considered in a parametric study. The results of this study are summarized in Table 2-14.

Table 2-14. Comparison of Different Approaches to Diversity Omnidirectional Systems

Type of Diversity	Vehicle Implemention	Changes in Den	Remarks
Spatial Direction	Separate receiver for each hemisphere, switch transmitter to better hemisphere.	None	Extra receiver, control logic and transmitter output switch required compared to other systems.
Time	Switch system alternately from one hemisphere to the other	None	Phase transients caused by switching interfere with phase lock. 3 db loss in transmit signal compared to No. 1.
Polarization	RHCP is one hemisphere, LHCP is the other	Require polarization diversity reception, switched transmission.	3 db loss in transmit signal compared to No. 1.
Frequency	Separate frequency channel reception and transmission for each hemisphere.	Dual channel reception and transmission required.	Doubles complexity of both vehicle and ground station system.

Variations in antenna design are shown parametrically in Table 2-15.

Table 2-15. Comparison of Various Designs of Antennas for Hemispherical Coverage

Antenna Type	Gain Variation Over Hemisphere (estimated db)	Feed Network	Remarks
Two arm conical spiral	7.5	Balun	Axial ratio 4 db max.
Four arm conical spiral	3	4-phase hybrid ring	Axial ratio 2 db max.
Eight arm conical spiral	2	8-phase hybrid ring	Axial ratio 1.6 db max. Losses slightly higher than four arm. Feed more complicated.
Turnstile over shaped ground plane	6	4-phase hybrid	Linear polarized in four directions
Cup turnstile	9	4-phase hybrid	Linear polarized in four directions

2.3.4 Transmitter

The transmitter includes a baseband assembly unit, modulator-exciters, and power amplifiers. The baseband assembly unit combines the 1.024 MHz engineering data subcarrier with the multiplexed video and high rate telemetry data or the ranging code depending upon the selection made and adjusts the relative amplitudes of the signal to an appropriate drive level for the phase modulators. The baseband assembly output drives three modulator-exciters in parallel. Two of the exciter outputs separately drive two travelling wave tube power amplifiers.

Three modulated carriers, two from the power amplifiers and one from the low power transmitter, are fed to the circulator switch assembly, which consists of five individual switches. The switches are arranged to enable any of the three transmitters to be connected to any of the four antennas via diplexers. The selection of the transmitting path is controlled by the transmitter selector, which derives its logic conditions from output power monitors and command sequences. The transmitter can be operated in either coherent or noncoherent modes. Lack of a coherent reference within the selected transmitter. Details of the components of the transmitter are given in the preliminary specifications of Figures 2-23, 2-24, and 2-25.

PRELIMINARY SPECIFICATION

Baseband Assembly

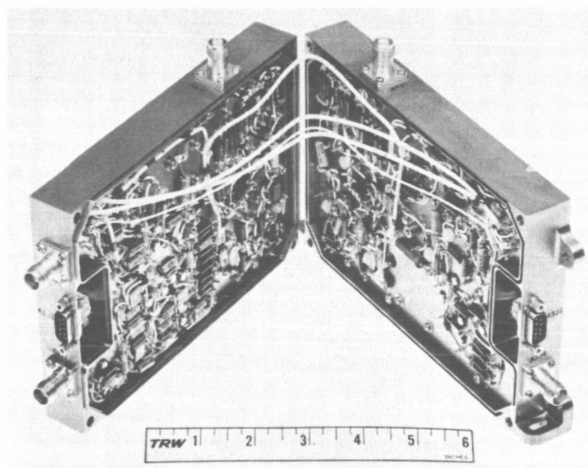
Purpose

Accepts multiplexed signals (video and telemetry) from the telemetry subsystem and turnaround ranging code from the receiver, sums the selected signals in an amplitude ratio which provides the correct modulation indices, and filters the signal to limit the baseband frequency spectrum. A ranging ON/OFF switch is included so that selection between video and ranging signal can be achieved.

Performance Characteristics

UNIT SPECIFICATION

1) Frequency range:	100 Hz 1.5 MHz ± 1.0 db -30 db a 3 MHz
2) Voltage gain:	Adjustable -10 to +10 db
3) Input/output impedance:	50 ohm
4) Maximum output swing pp:	3.0 V pp minimum
5) Individual level control:	10 db
6) Range:	
7) Operating temperature:	0°C to +65°C
8) Weight:	1.0 lb maximum
9) Dimensions:	3.5 x 5.0 x 2.5 in. maximum



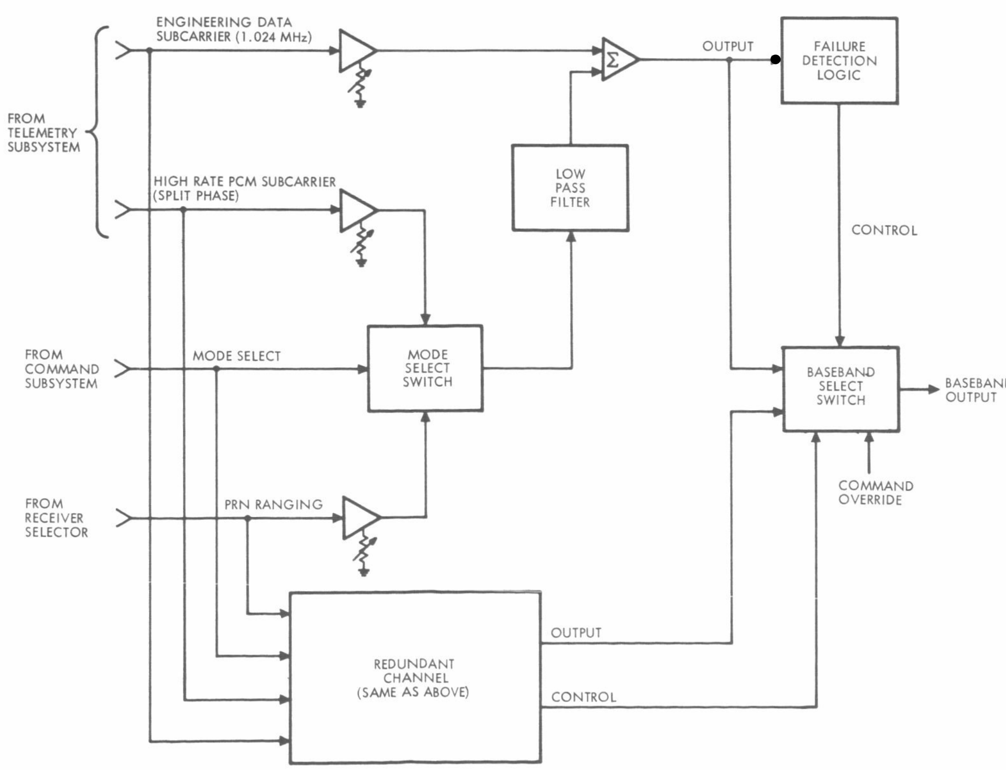


Figure 2-23

PRELIMINARY SPECIFICATION

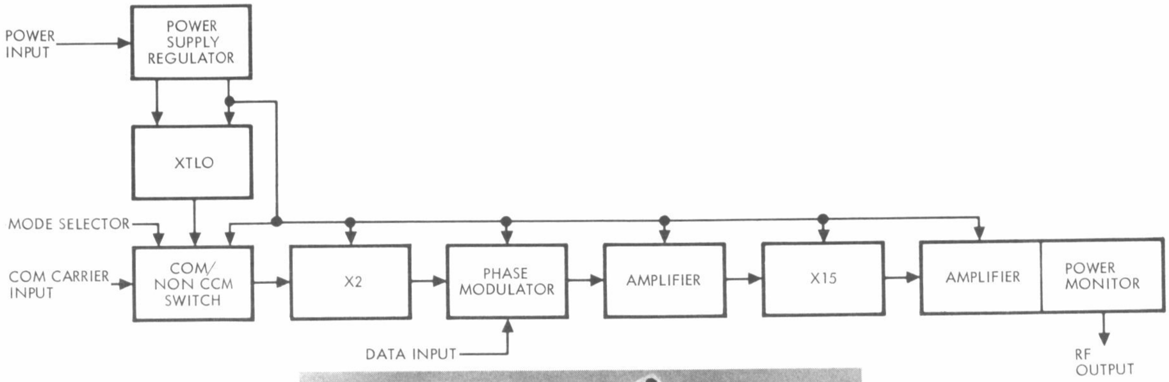
Modulator Exciter

Purpose

The modulator-exciter phase modulated the telemetry and ranging signals onto an S-band carrier operating in either a coherent or noncoherent mode. For the coherent mode, the coherent/noncoherent switch selects the coherent source input from the receiver. Otherwise, the output from an internal crystal oscillator is selected. The baseband signal is linearily phase-modulated onto the carrier, then multiplied and amplified to a level of 100 mw for the modulator-exciter and 1.0 watt for the low-power transmitter

Performance Characteristics

Coherent input frequency:	70 MHz (approximately)
Output frequency:	30 times the input frequency, approximately 2295 MHz
Noncoherent reference source:	70 MHz (approximately)
Power output:	100 mw minimum (exciter 1.0 watt minimum (low power transmitter)
Output phase stability (noncoherent mode):	6 deg peak maximum
Output frequency stability of the reference source:	a) short-term 10^{-7} in 20 min b) long-term, 10^{-6} in 12 hr
Modulation bandwidth:	2.0 MHz
Peak modulation:	3.0 rads
Modulation sensitivity:	0.75 rad/volt
Modulation input impedance:	5.0 kilohm shunted by 25 pf maximum
Input/output impedance:	50 ohm
VSWR input/output:	1.2:1
Spurious outputs:	all spurious output shall be less than 50 db below unmodulated carrier
Weight:	3.0 lb
Physical dimensions:	8.25 x 5 x 1.65 in.
Operating temperature:	0°C to +60°C



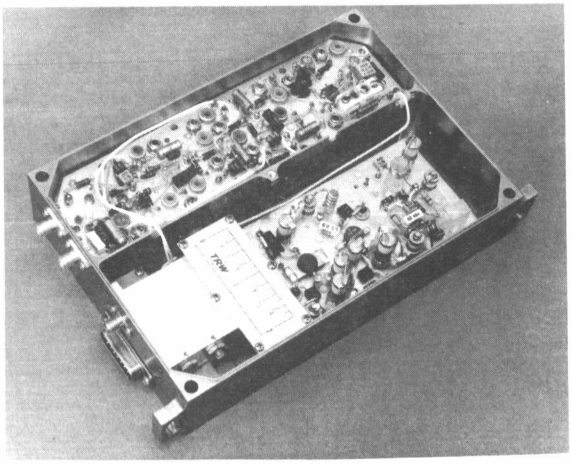


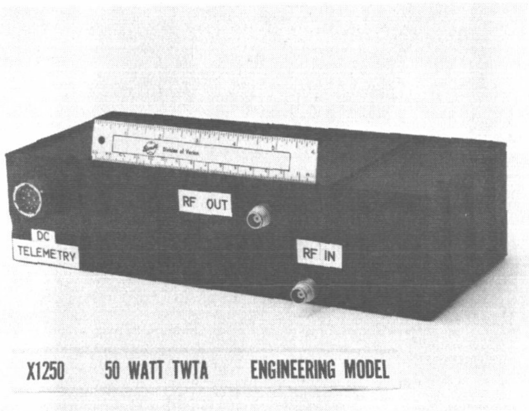
Figure 2-24

PRELIMINARY SPECIFICATION

Power Amplifier (TWT) and Power Supply

Purpose

The power amplifiers raise the S-band signal from the modulator-exciters to a level of 50 watts. The power amplifier includes a traveling wave tube, power supply, calibrated telemetry voltage generation circuits, RF power monitor, and RF output filter. A unit in compliance with the specifications given below is in a pilot production stage.



Performance Characteristics

UNIT SPECIFICATION

1) Frequency:	2295 ± 5 MHz (1 db points)
2) Power output:	50 watts minimum
3) RF gain saturated:	30 db minimum
4) Input/output impedance:	50 ohms
5) VSWR input/output:	1.2:1
6) Noise figure:	35 db maximum
7) Power supply:	+37 volts to +50 VDC
8) Overall RF/DC effeciency:	30 percent minimum
9) Spurious output:	Not more than -80 dbm in any 1 cps band in the range 2290 to 2300 MHz
10) Harmonics:	Not more than -60 db below unmodulated carrier
11) Turn on time:	90 sec maximum
12) The TWT and power supply is not susceptible to electromagnetic radiation and does not generate more than 0.2 gamma at 6 feet from the gun end.	
13) Operating life time:	10,000 hours minimum continuous without performance degradation
14) Operating temperature range:	-30°C to +100°C (baseplate temperature)
15) Vibration (operating):	20 G's 20 to 2,000 cps
Shock (operating):	200 G's 1 ± 0.5 msec
16) Weight (TWT and power supply):	7.8 lb
17) Physical dimensions:	12 x 6 x 3 in.

Figure 2-25

2.3.5 Receivers

The receiving equipment includes preamplifiers and phase lock receivers. Specifications for these units are given in Figures 2-26 and 2-27.

2.3.6 Selectors and RF Switches and Diplexer

The selectors and radio frequency switches include the low gain antenna selector, the transmitter selector, the receiver selector, and the circulator switches. Specifications for these units are given in Figures 2-28, 2-29, 2-30, 2-31, and 2-32.

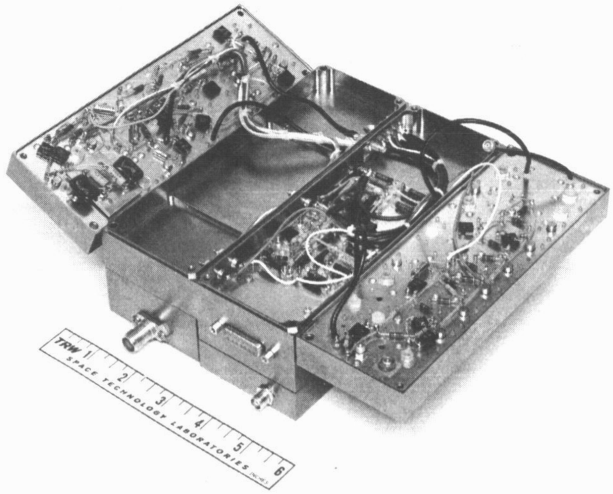
PRELIMINARY SPECIFICATION

Phase-Lock Receiver

Purpose

The phase-lock receiver provides a coherent carrier reference and a demodulated baseband signal. It also provides a "signal present" output proportional to the received signal strength.

The receiver is a double hetrodyne unit with the second intermediate frequency operating at the same frequency as locally generated reference oscillator. Two phase detectors demodulate the second IF. The PRN ranging is obtained from a wideband phase detector and the command subcarrier from a narrow band phase detector which also produces the error signal to drive the phase locked first local oscillator. The error signal is filtered and applied to a VCO. The controlled output of the VCO is multiplied and combined with the reference oscillator to produce the phase-locked first and second local oscillator frequencies. The "signal present" output is obtained by correlating the reference oscillator output with the second intermediate frequency signal in a coherent amplitude detector. (CAD)



Performance Characteristics

1) RF input frequency:	2115 MHz ± 5 MHz
2) Carrier tracking loop noise bandwidth:	32 Hz
3) Noise figure:	10 db maximum
4) IF bandwidth:	4.0 MHz
5) Video channel bandwidth:	1.5 MHz
6) Operating dynamic range:	-145 to -30 dbm
7) Phase-lock loop threshold:	145 dbm (6 db SNR)
8) Video output:	0.2 V pp to be squelched when the receiver is not locked
9) Static phase error:	Output to be provided to indicate loop stress
10) Signal present output:	To be provided to indicate lock-on condition
11) Coherent carrier reference:	70 MHz output for coherent drive reference which is 8/221 of the RF input frequency
12) RF input/output impedance:	50 ohms
13) RF input/output VSWR:	1.4:1
14) Weight:	3.0 lb
15) Dimensions:	7.25 x 5 x 3 in.
16) Operating temperature:	0°C to +65°C

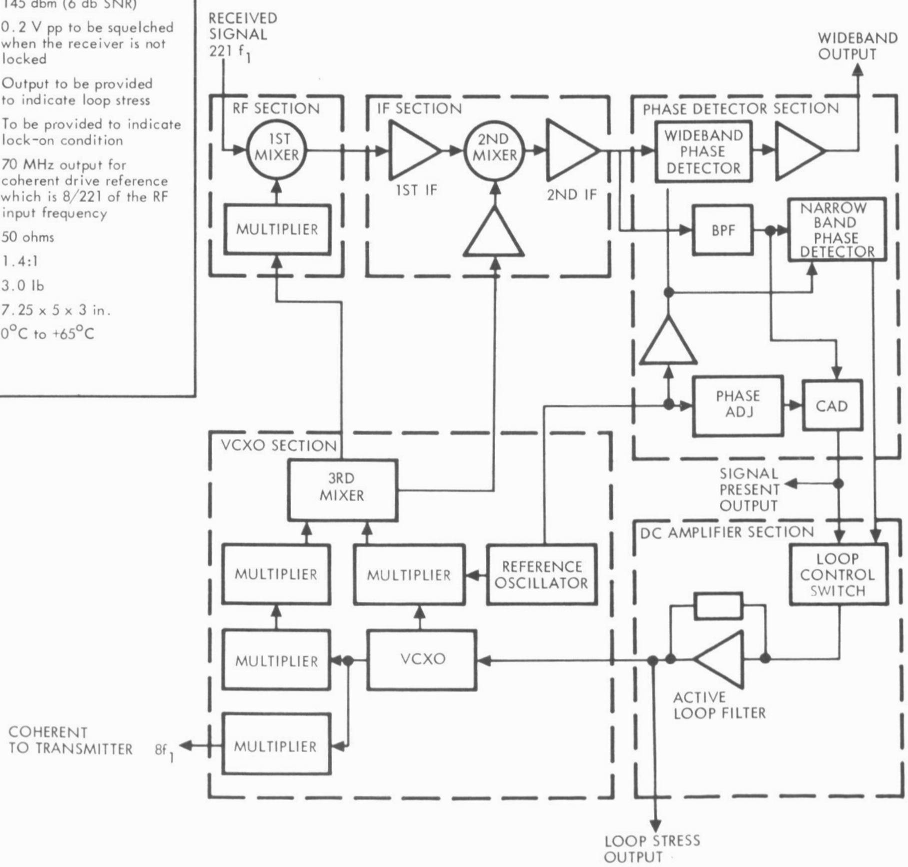


Figure 2-26

PRELIMINARY SPECIFICATION

S-Band Preamplifier

Purpose

The preamplifier receives the low-level S-band signal from the antenna and provides low-noise wideband amplification for the phase-lock receiver.

The unit is an all solid state four-stage common-emitter amplifier built on a strip line board. The first stage matches the input impedance and is optimized for minimum noise figure and maximum gain. The two intermediate stages provide necessary gain, while the output stage provides required output drive and matches with the load.

Performance Characteristics

Frequency:	2115 MHz ± 10 MHz
Gain:	20 db nominal
Noise figure:	5 db maximum
Power output:	-10 dbm
Input/output impedance:	50 dbm
Input/output VSWR:	1.4:1 maximum
Supply voltage:	-15 vdc ± 5%

Physical Characteristics

Weight:	0.4 lb max
Dimensions:	3 x 2 x 1 in. max
Operating temperature range:	0°C +60°C

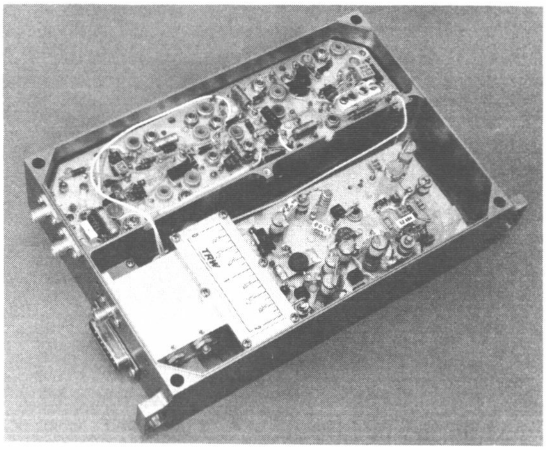


Figure 2-27

PRELIMINARY SPECIFICATION

Low Gain Antenna-Selector

Purpose

The low gain antenna-receiver selector combines the command subcarriers outputs from the two receivers connected to the low gain antennas. Switches at the inputs to the combiner are controlled by a threshold detector which inhibits the output of a receiver if the signal present voltage falls below a preset level. The unit also selects the appropriate PRN ranging signal and coherent carrier reference based on whichever receiver has the highest received signal level. These signals are transferred to the receiver selector for subsequent processing.

Performance Characteristics

CONTROL INPUTS

Signal present (2):	Analog DC voltage proportional to the received signal strength

SIGNAL INPUTS

Baseband signals (2):	PRN ranging code and command subcarrier
Coherent carrier reference (2):	71 MHz sinusoidal signal

SIGNAL OUTPUTS

Baseband signal:	PRN ranging code and command subcarrier to receiver selector
Coherent carrier reference:	71 MHz sinusoidal signal to receiver selector
Control signal:	Selects low gain antenna for downlink transmission

Physical Characteristics

Weight:	0.3 lb
Physical dimensions:	3.5 x 5 x 1.5 in.
Operating temperature:	0°C to +60°C

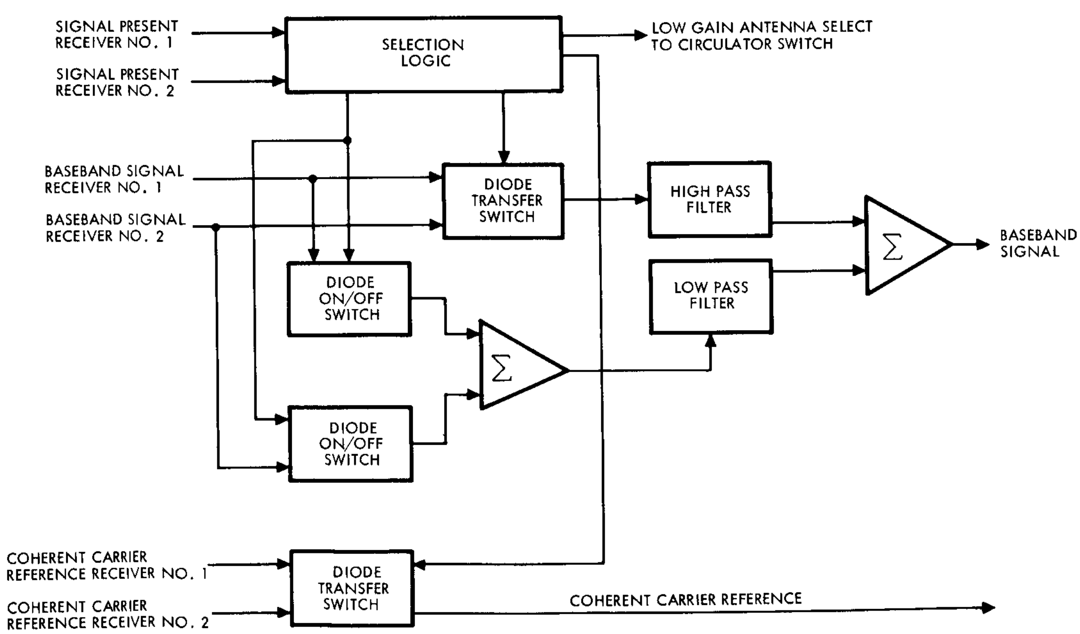


Figure 2-28

PRELIMINARY SPECIFICATION

Transmitter Selector

Purpose

The transmitter selector selects one of three transmitters and either a coherent or a noncoherent operating mode, under control of preset logic or external commands. To control transmitter operating modes, it accepts and translates analog inputs from four receivers. If any one of the four receivers is in-lock, the output from the OR-logic enables the transmitters to operate in coherent mode. Through the logic circuits, the input from the command sequencer generally selects one of the three transmitters. However, the output monitor signals from the three transmitters are used to assure that the selected transmitter is operating. It it fails, the logic circuitry will automatically switch in another transmitter.

Performance Characteristics

INPUTS:

Power monitor inputs (3):	Analog DC voltage proportional to transmitter power
Receiver in-lock inputs (4):	Analog DC voltage proportional to received carrier power
Command sequencer inputs:	Digital, format to be specified

OUTPUTS:

Transmitter enable:	Three outputs of which only one is energized. Each output is capable of switching the corresponding transmitter
Antenna select:	Transfer antenna select command to circulator switch
Coherent/noncoherent selection:	Capable of driving three switches in parallel

Physical Characteristics

Operating temperature:	0°C + 65°C
Weight:	0.9 lb
Dimensions:	3.5 x 5 x 1.62 in.

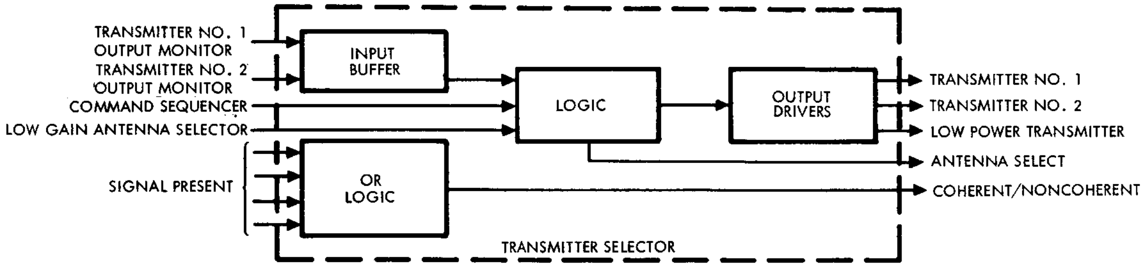


Figure 2-29

PRELIMINARY SPECIFICATION

Receiver Selector

Purpose

The receiver selector selects the outputs from one of the four phase-lock receivers, based on preset logic and command inputs. The logic circuit is similar to that of the transmitter selector. The phase-lock receiver associated with the antenna currently employed for downlink transmission has the first priority. Other receivers are used in turn on a relative signal strength basis when this receiver is not in-lock.. The actual selection is performed by the diode switches in the receiver selector. The ranging code and the coherent carrier reference are buffered with amplifiers before delivery to the baseband assembly unit and transmitter selector. The command subcarrier passes through a low pass filter and buffer amplifier before it is sent to the command decoders.

Performance Characteristics

SIGNAL INPUTS:

Coherent carrier references (3):	71 MHz sinusoidal signals, one from each of the three transmitters
Baseband signals (3):	Demodulated ranging codes and command subcarriers, one from each of the three recievers

CONTROL INPUTS:

Command sequencer:	Digital input, format to be specified
Signal present inputs (4):	Analog signal from the receivers. It is proportional to the received carrier level

SIGNAL OUTPUTS:

Ranging code:	Demodulated PRN ranging code to baseband assembly
Command subcarrier:	Demodulated command subcarrier to command decoder
Coherent carrier reference:	71 MHz sinusoidal signal to modulator exciter

Physical Characteristics

Operating temperature:	0°C TO + 65°C
Weight:	0.9 lb maximum
Dimensions:	3.5 x 5 x 1.62 in. maximum

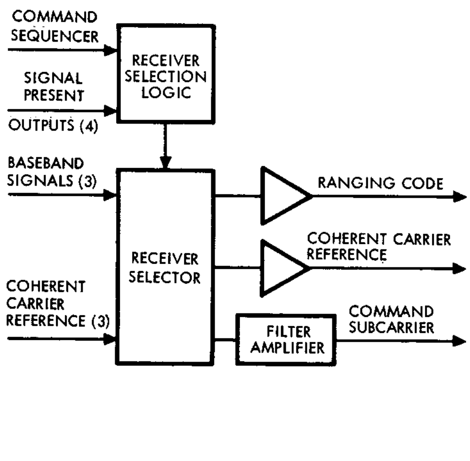


Figure 2-30

PRELIMINARY SPECIFICATION

Circulator Switch Assembly

Purpose

The circulator switch assembly transfers any of the three transmitter outputs to any of the four antennas, via diplexers. It consists of five independent circulator switches which are controlled independently. Switching is accomplished by controlling the state of circulator polarization. The switches and the control circuitry are integrated into one package.

The unit exhibits full latching properties, requiring no power to maintain a particular polarization. It requires a high energy pulse to change the state, approximately one ampere for 50 microseconds. The control circuitry provides the required drive. Units meeting all of the Voyager requirements are readily obtainable.

Performance Characteristics

Frequency:	2295 MHz ± 5 MHz
Insertion loss PRN section:	0.3 db maximum
Isolation between sections:	25 db minimum
Power handling capacility:	150 watts cw
Input/output impedance:	50 ohms
Input/output VSWR (passbands):	1.2:1 maximum
Power supply:	+37 to 50 VDC
Control logic inputs:	+1.0 V pp DC

Physical Characteristics

Weight:	7.5 lb
Dimensions:	10 x 6 x 6 in.
Temperature range (operating):	0°C to +65°C

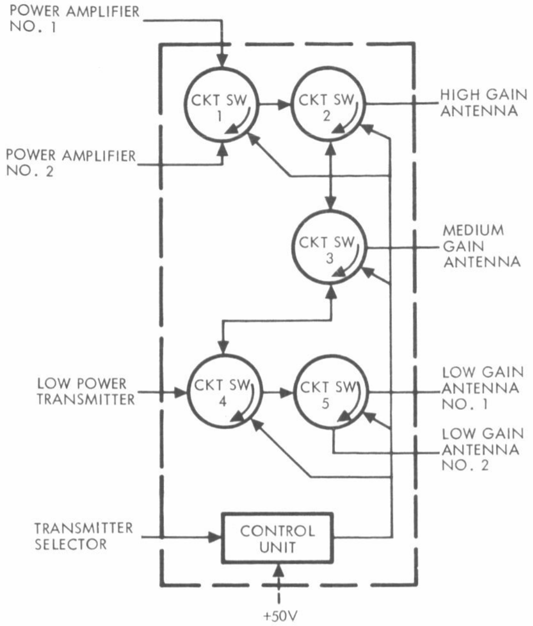


Figure 2-31

PRELIMINARY SPECIFICATION

Diplexer

Purpose

The diplexer allows a common antenna to be used for both transmission and reception simultaneously, by directing the transmitted and received signal to the antenna and receiver respectively.

The unit effectively consists of two bandpass filters sections and a matching junction. The unit is readily available from several suppliers.

Performance Characteristics

Frequency:	2295 MHz ± 5 MHz transmit 2115 MHz ± 5 MHz receive
Passband insertion loss:	0.3 maximum for receive/transmit
Isolation:	80 db minimum at both transmit and receive frequencies
Input/output impedance:	50 ohms
Input/output VSWR (passband):	1.2:1 maximum
Power-handling capability:	150 watts cw
Weight:	1.0 lb maximum
Dimensions:	7.5 x 4 x 2.5 in. maximum
Operating temperature:	+0°C to +65°C

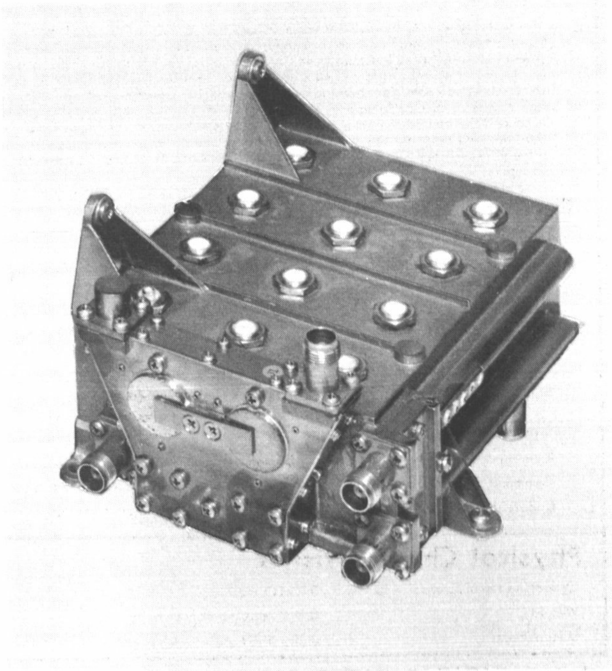


Figure 2-32

3. COMMAND SUBSYSTEM

3.1 SUMMARY

The command subsystem performs demodulation and synchronization of command signals from the S-band radio subsystem and the decoding of the resulting digital commands. The outputs are fed to the spacecraft subsystems or to the capsule as directed by addresses in the command data. A simplified block diagram of the command subsystem is shown in Figure 3-1. The location of the subsystem in the spacecraft is shown in Figure 3-2.

The command subsystem contains two bit synchronizers and two decoders configured for cross-strapped redundancy. Both bit synchronizers operate in parallel and command word bits are utilized to select decoders.

The decoders receive data bits, bit synchronization pulses, and lock signals from the demodulator/bit synchronizers. The decoder performs command word format and parity checks, and in response to routing addresses transmits serial command word bits to the computer and sequencer or directly to other spacecraft subsystems and provides direct discrete commands on an immediate basis to other spacecraft subsystems.

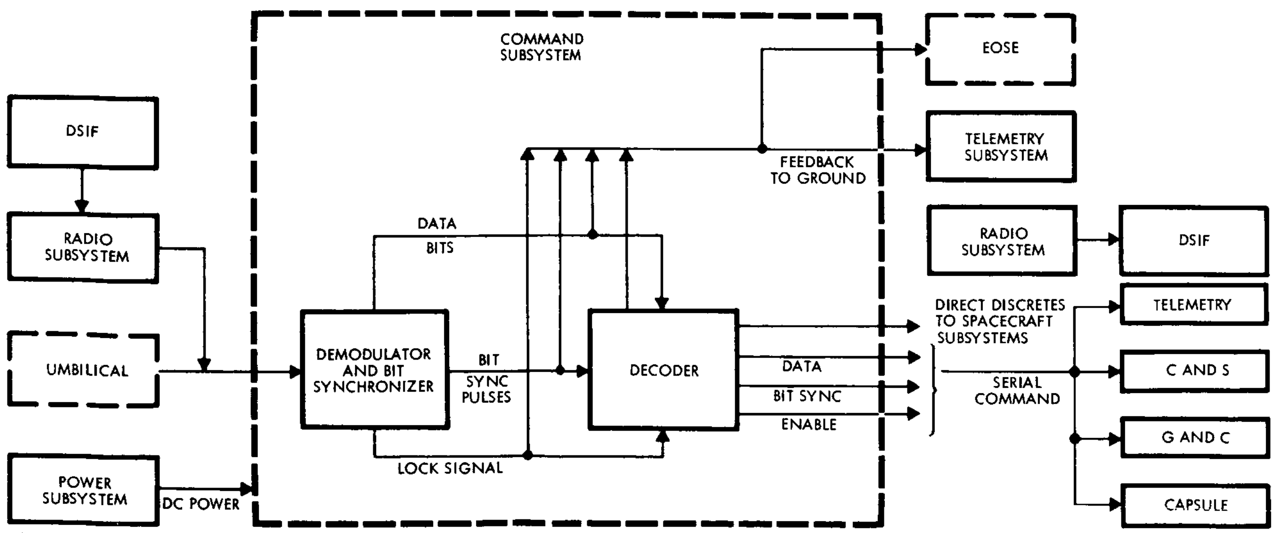


Figure 3-1
THE COMMAND SUBSYSTEM is shown above in a simplified block diagram form. Included are the interfaces, with the rest of the spacecraft, the capsule and Earth stations.

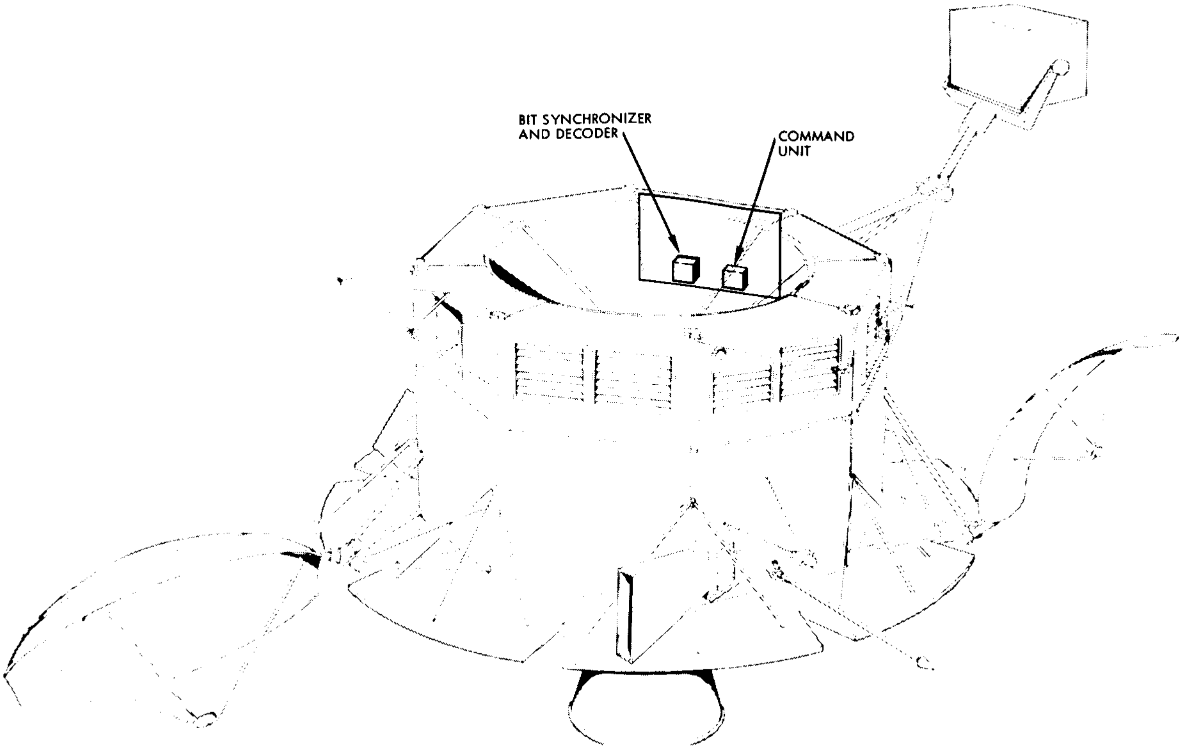


Figure 3-2
THE COMMAND SUBSYSTEM is located on one panel of the recommended spacecraft.

3.2 SUBSYSTEM FUNCTIONAL DESCRIPTION AND PERFORMANCE

The demodulator/bit synchronizer receives inputs from the S-band radio subsystem. While on the launch pad it can receive inputs by hard-line through the umbilical connection.

The decoder is capable of processing and decoding up to 256 discrete commands. In addition, it processes serial messages and feeds them to 12 destinations, including:

- 12-bit serial words to each of three antenna command registers
- 32-bit serial words to each of nine addresses, consisting of two computer and sequencer command input registers (primary and backup sequencers), six capsule registers and one telemetry register.

Preliminary specifications of the command subsystem demodulator/bit synchronizer are shown in Figure 3-3. Those for the decoder are shown in Figure 3-4. The synchronization acquisition time of the command

PRELIMINARY SPECIFICATION

Demodulator/Bit Synchronizer

Purpose

- Established bit synchronization and demodulates command signals from the radio subsystem.
- Provides digital command bits, clock signals and a sync lock indication signal to the decoder.

Physical Characteristics

- Volume: 50 cu in.
- Dimensions: 7 x 7 x 1 in.
- Weight: 2.6 lb

Performance Characteristics

(for one unit; two required to provide redundancy)

Parameter	Specification
• Input signal	subcarrier, biphase modulated with NRZ-M code
• Bit rate	8 bits/sec
• Tracking range	0.1% of the subcarrier rate
• Capture range	0.1% of the subcarrier rate
• Input subcarrier frequency stability	± 0.05%
• Bit error rate (BER)	BER vs SNR will be within 1.8 db of the theoretical to a SNR corresponding to BER of 10^{-6}
• Acquisition time	2 minutes maximum
• Output signal	NRZ-C (NRZ-L)
• Sync status output	An output will be provided to indicate bit sync status
• Primary voltage	28 ± 4 VDC
• Power	3 Watts

Figure 3-3

PRELIMINARY SPECIFICATION

Decoder

Purpose

- Decodes digital commands received from the bit synchronizer and provides outputs to the spacecraft subsystems or to the capsule as directed by instructions in the command data.

Outputs:

- Direct discrete orders
- Serial command words

Performance Characteristics

(for one unit: two required to provide redundancy)

- Maximum number of external address for serial command words: 13
- Maximum number of direct discrete orders: 256

Physical Characteristics

(For two redundant assemblies)

- Volume: 80 $in.^3$
- Dimensions: 7 x 7 x 1.63 in.
- Weight: 5.3 lb
- Power: 3 watts

Figure 3-4

link is minimized, and the bit error rate (BER) versus the signal-to-noise-ratio (SNR) is within 1.8 decibels of the theoretical curve of an SNR corresponding to a BER of 10^{-6}. The probability of generating an erroneous command, attributed to bit errors and false lock indication, will be less than 10^{-8}. The probability of rejecting a valid command, attributed to bit errors and false lock, will be less than 10^{-4}.

The command subsystem decoder contains three pairs of assemblies: the input selector, the input decoder, and the output decoder. These are illustrated in the functional block diagram shown in Figure 3-5. The input decoder processes incoming messages according to their formats.

Two types of command message formats are received. Message format 1 is used for transmitting direct discrete commands that will be generated by the output decoder. Message format 2 is used for transmitting serial commands to other spacecraft subsystems including the capsule, computer and sequencer, and guidance and control subsystem.

There is an element in the command word which is not part of the command but is required for synchronization. Once a command word has been received in an acceptable manner, other command words can be received without this element, providing there is no gap between words. After a) first establishing bit synchronization, b) detecting a parity error, or c) detecting a false decoder or subsystem address, it is necessary to again receive the element before a command will be accepted. The command word contains additional elements defining addresses, parity, and

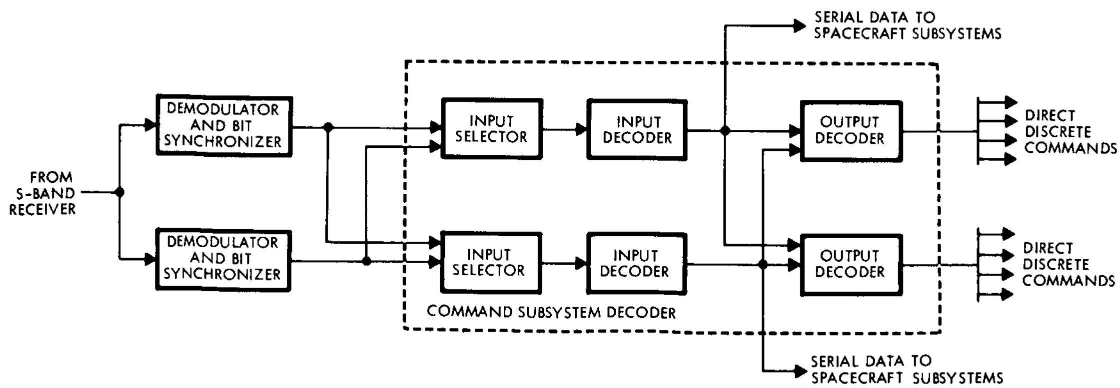


Figure 3-5

THE FUNCTIONAL BLOCK DIAGRAM OF THE COMMAND SUBSYSTEM shows the decoder broken down into a input selector, input decoder, and output decoder. The diagram identifies also the full redundancy and the cross-strapping between individual units.

message formats of the incoming data as well as interword synchronization. These elements are checked for validity in the input decoder prior to the generation of enable signals.

The command word structure and the subcarrier frequency are classified according to JPL Document EPD-20, Addendum H. In order to prevent classification of this report, these two parameters are not listed herein and can be found by referring to TRW Systems Phase 1A, Task B Final Technical Report, Volume 2, Section II, page 228.

A summary of all the decoder outputs is shown in Table 3-1. The enable signals are generated on a selective basis after the entire input command message has been checked.

The direct discretes are issued when the indicated output decoder receives the enable. Many of the discretes perform the same function as discretes generated by stored program command in the computer and sequencer. These discretes from the command subsystem decoder serve as backup to those of the computer and sequencer. A list of the functions performed by the command subsystem decoder is presented in Table 3-2 at the end of this section.

The parts count assessments yield failure rates for the units are as follows:

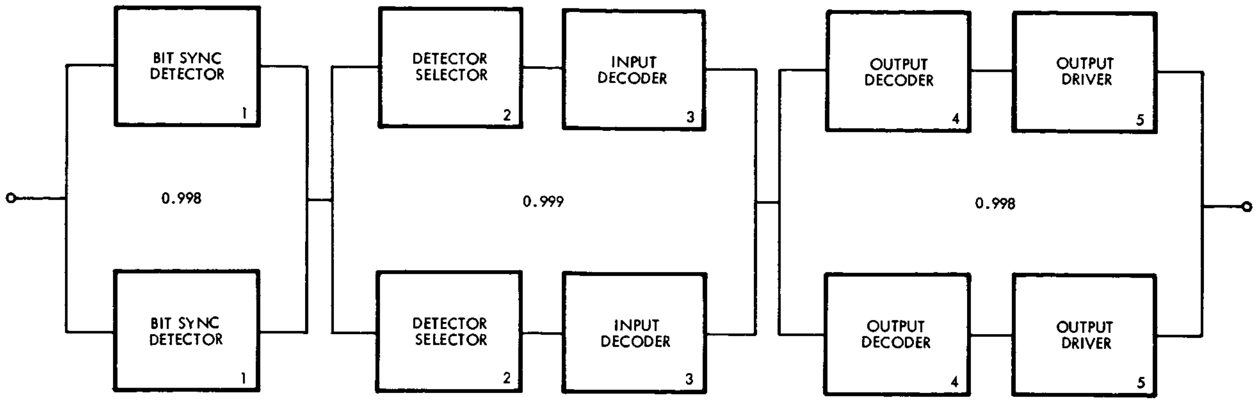
	Failure Rate/10^9 Hours
Demodulator/synchronizer	9417
Decoder input selector	300
Input decoder	2250
Output decoder	6000

The reliability estimate was made for a mission time of 6800 hours for the configuration shown in Figure 3-6. The command unit reliability is estimated at 0.995.

Table 3-1. Command Subsystem Decoder Outputs

The direct discretes are activated on an immediate basis from the uplink data and are generated independently from discretes commands of the Computer and Sequencer Subsystem. However, the direct discretes serve primarily as backup to the stored program commands of the Computer and Sequencer Subsystem.

- Enable signal to C and S command input register (primary)
- Enable signal to C and S command input register (backup)
- Enable signal to G and C medium-gain antenna hinge angle register
- Enable signal to capsule (6 lines)
- Enable signal to telemetry
- Enable signal to other output decoder
- Bit sync signal (separate line to each recipient of the above signals)
- Serial bit stream output (common line to all recipients of above enable signals)
- Address gate state selecting other output decoder
- Direct discrete orders (up to 256 separate lines)
- Composite enable signal to telemetry
- Telemetry buffer (12 bits)
- DC power voltage (2) (to telemetry)



$$R_{COMMAND} = \left[1 - (1 - R_1)^2\right]\left[R_2R_3(2 - R_2R_3)\right]\left[R_4R_5(2 - R_4R_5)\right]$$
$$= 0.995$$

Figure 3-6
RELIABILITY OF THE COMMAND SUBSYSTEM is estimated as 0.995 on the basis of the above block diagram. Detailed calculations are presented in Volume 4, Section 3.8.5

3.3 COMPONENT DESCRIPTION

3.3.1 Demodulator/Bit Synchronizer

The subcarrier demodulator/bit synchronizer recommended for use in the command subsystem is an improvement over the Mariner C demodulator/synchronizer proposed in the Task B report of TRW. This new unit is currently being developed by TRW under NASA Contract No. NAS 9-7061. The unit design is an adaptation of concepts previously used by TRW on a similar unit for the Pioneer Interplanetary Spacecraft Program. The Pioneer unit exceeded its performance specifications. The performance requirements for this unit are similar.

The tabulation below compares the newly proposed subcarrier demodulator/bit synchronizer with the previously considered Mariner C unit.

Parameter	Mariner C	Recommended Demodulator/Bit Synchronizer
Acquisition time	20 minutes	2 minutes or less
Bit rate	1 bit/sec	8 bits/sec, the link data rate is increased by a factor of 8, and results in a negligible loss in power margin of only 0.6 decibels. The reason for this is that no power is used for synchronization; it is derived from the data.
Complexity	Relatively complicated because of the elaborate synchronization scheme required.	Less complicated; does not require an elaborate synchronization scheme.
Design flexibility for increased bit rates	Essentially requires an additional demodulator/synchronizer for each additional bit rate desired.	Relatively simple modification required for additional bit rates.

The basic problem to be solved in demodulation of the digital information is to construct a matched filter for the input wave form over the bit time T. An optimum filter for the biphase information in the presence of white noise is shown in Figure 3-7. The input S(t) is multiplied by a locally generated reference g(t) which is maintained in frequency and phase with the subcarrier. The output of the multiplier is passed to an integrate and dump filter which is reset at each bit transition by a locally generated dump signal. The output of the integrator which is sampled just prior to reset contains the best estimate of the polarity of the bit information over the time 0 to T. The modulating NRZ-M data might, dependent on polarity consideration, be as shown in Figure 3-7. The reconstructed NRZ-L data will approximate the modulating data with a one-bit time delay.

No separate synchronization information will be transmitted for the demodulator, so the reference g(t) and the bit sync information for generating the dump signal must be extracted from the input S(t). This is accomplished by use of a modification of the Costas Synchrolink or I-Q demodulator.

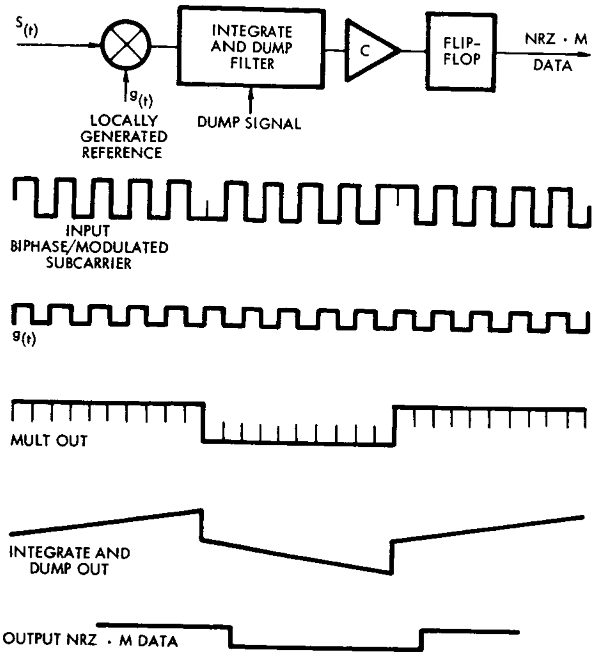


Figure 3-7

A MATCHED FILTER for $S_{(t)}$ as shown above is the principle element of the bit synchronizer. H represents an optimum filter for biphase information in the presence of white noise. Below the block diagrams are the pertinent wave forms. The reconstructed NRZ-L data output approximates modulating data with a one bit time delay.

A block diagram of the recommended demodulator/synchronizer is shown in Figure 3-8. The references for the I and Q multipliers in the demodulator loop are the carrier frequency in-phase and shifted 90°. The output of each multiplier is passed through an integrate and dump filter which, when in synchronism, is reset at the bit time. The output of the I data filter is impressed on a high-resolution comparator C which continuously decides the polarity of the filter output. The output of the comparator is sampled by the data flip-flop at the bit time (provided by the bit sync loop) after maximum averaging in the filter. The code converter changes the NRZ-M code to NRZ-L code.

The demodulator Q channel provides loop sense information to the voltage controlled crystal oscillator (VCXO). The output of the Q integrate-and-dump filter is sampled and held for the succeeding bit period. This sample, which is the VCXO correction, is multiplied by the NRZ-M data to remove sense ambiguities due to information modulation and is passed through the Q loop filter to the VCXO.

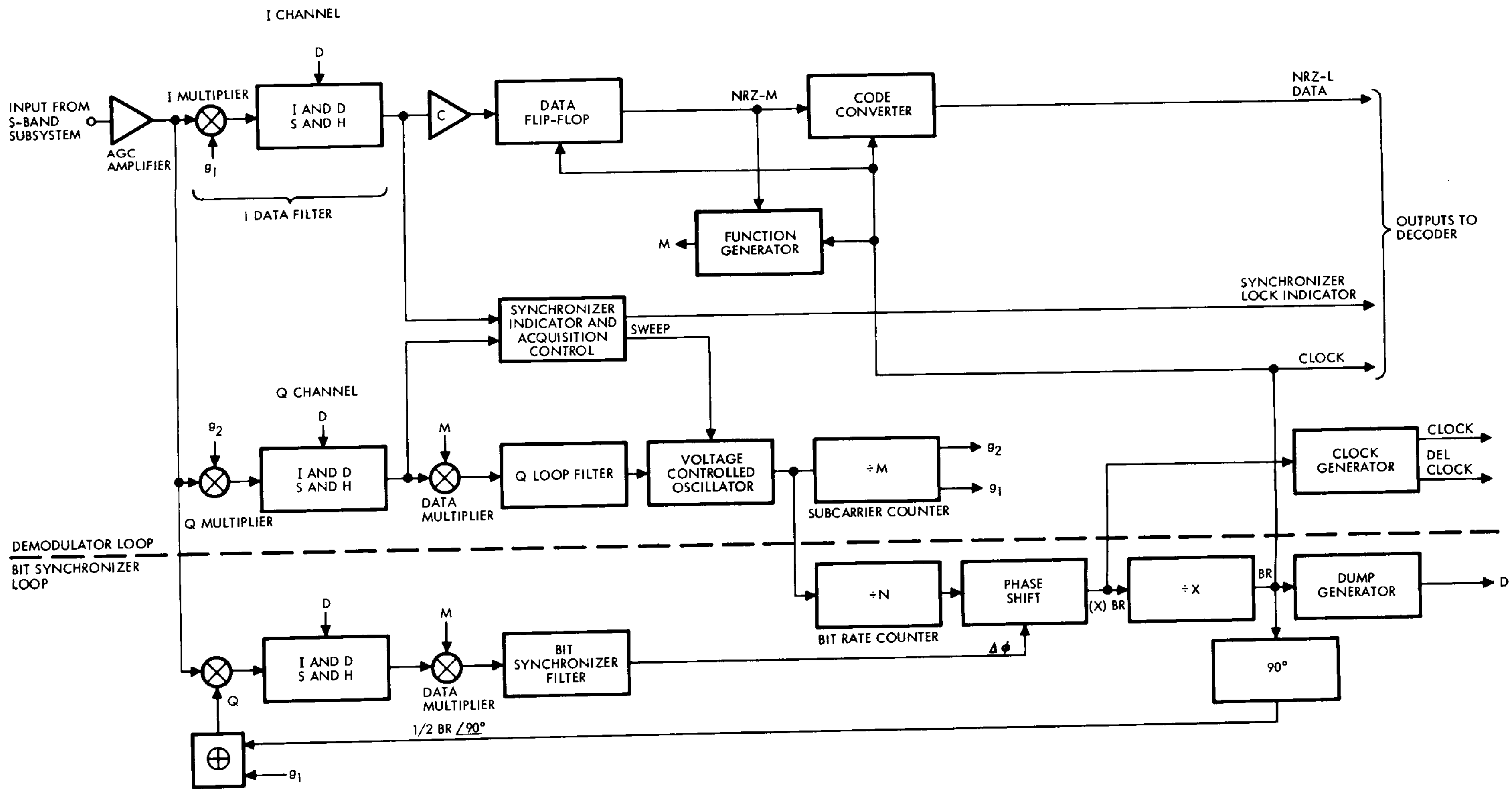


Figure 3-8
The Demodulator/Synchronizer uses the in-phase carrier frequency and the carrier frequency shifted by 90° as references for the I and Q multipliers. The output of each multiplier is passed through an integrate and dump filler which, when in synchronism, is reset at the bit time.

The bit synchronizer consists of a Q channel which shares the demodulator I channel for data multiplication. The reference for the bit sync Q channel (Q') is a signal which is the "exclusive OR" of the demodulator I channel reference and half the bit rate shifted 90 degrees. The carrier component of this signal redemodulates the carrier, and the bit rate component allows bit synchronization. The output of the demodulator I multiplier could, of course, be multiplied by the bit rate component to achieve the same result, but this would require two multiplier designs. The bit sync multiplier would require essentially DC coupling when operating at low bit rates with low transition densities. Delays resulting from cascaded multipliers are also undesirable. The output of the bit sync multiplier is filtered, sampled, and held, multiplied by the NRZ-M data to remove sense ambiguities, and applied ($\Delta\phi$) to the phase-shift circuit. The phase-shift circuit uses the $\Delta\phi$ correction signal to shift the phase of the output from the bit rate counter relative to the subcarrier. Bit sync is attained when the output of the phase-shift circuit is in phase with the data transitions in the incoming information. The XBR output of the phase-shift circuit serves as the clock signal from the clock generator, code converter, function and dump generators.

Performance and optimization of the acquisition task requires sensing the status of the I and Q loops to determine when the magnitude of the I is maximum and the Q is minimum. A condition is also imposed by the bit sync loop such that the ratio of I/Q is optimally maximum when bit sync lock is achieved.

Determination of the status of the loops under conditions of poor SNR's requires that the I and Q information be sampled where it is cleanest. This constraint is best met at the output of the matched filter in each loop.

Acquisition and synchronization with the input signal is performed automatically by the unit. The sync indication and acquisition control circuitry averages the outputs of the matched filters in the I and Q loop and subtracts the averaged Q loop signal from the averaged I loop signal (I-Q). During acquisition the VCXO is swept through subcarrier phase null. As phase null is approached the I-Q function is maximized provided that the bit sync loop phase error is not at a maximum. Bit sync is attained when a comparison circuit senses the maximized I-Q signal and generates the sync lock signal which causes the VCXO to stop sweeping. If the unit should drop out of lock, acquisition control circuitry would automatically start sweeping the VCXO and the above process is repeated to obtain a lock indication.

3.3.2 Decoder Input Selector

The decoder input selector network first performs the selection of the command word bits and the bit sync pulses from the appropriate one of the redundant bit synchronizers as a function of their lock states and applies these signals to the input decoder with which it is associated. If only one of the bit synchronizers is in lock, the signals coming from it are used. If both bit synchronizers are in lock, the signals from the associated synchronizer are used. One of the redundant bit synchronizer units is considered as being associated with one of the redundant decoder units. Secondly, the network generates an out-of-lock reset signal as a function of the lock states of the bit synchronizers and their changes in states, and applies the signal to the input decoder. If the associated bit synchronizer is in lock and goes out of lock, the reset signal is generated. If the other bit synchronizer is the only one in lock and it goes out of lock, a reset signal is generated. If the associated bit synchronizer changes from not-locked to locked, the reset signal is generated.

3.3.3 Input Decoder

The input decoder provides the routing function to the various subsystems in response to information in the command word. The input decoder examines the command word to determine the validity of the message, establishes the appropriate destination to receive the data, and performs checks on the format parity. The input decoder has the modes and states corresponding to the command word formats.

A block diagram of the input decoder is shown in Figure 3-9. Bit sync pulses are used to step the bit time counter and shift the address register. The bit time counter provides timing signals at specific bit locations in the processing of the command message. The command word bits (CWB) are examined and routed according to the information contained in the message.

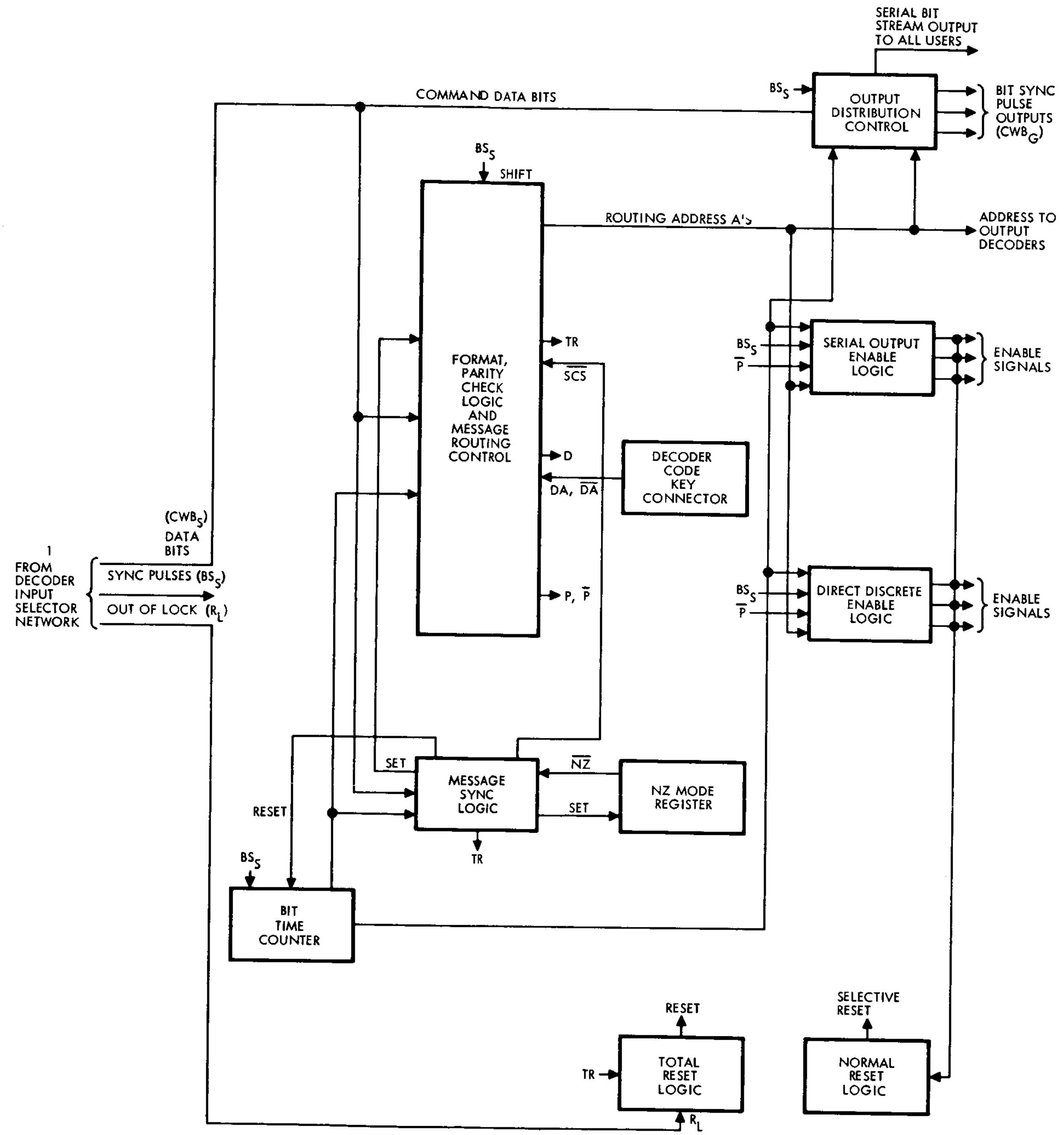


Figure 3-9

The INPUT DECODER uses bit sync pulses to provide the clock used to shift and check the format and parity of the command word data bits. The data is routed according to the information contained in the message.

The out-of-lock reset signal enters the total reset logic, causing a total input decoder reset. The NZ mode control, when reset, performs the premessage sync check. When the control is set, command message processing of the incoming message is enabled.

Each element of the message is examined as it enters the input decoder. The command word bits are steered through the input decoder blocks where checks are made on parity, format, and information content. At prescribed bit times, the bit sync pulses are sent to the output distribution gates and the command word bits are sent to all serial bit stream users, including the output decoders. The command word bits have no effect on the recipient unit unless it also receives bit sync pulses. The bit sync pulses are selectively routed to the particular recipient designated by elements in the message.

If all checks performed by the input decoder prove positive, an enable signal is sent to the message recipient signalling message validation. When the enable signal is generated, the input decoder is reset in readiness for the next message. If any check fails, a total reset of the input decoder occurs and the enable signal is not sent.

3.3.4 Output Decoder

A block diagram of the output decoder is shown in Figure 3-10. The incoming command word bits and bit sync pulses from the input decoder

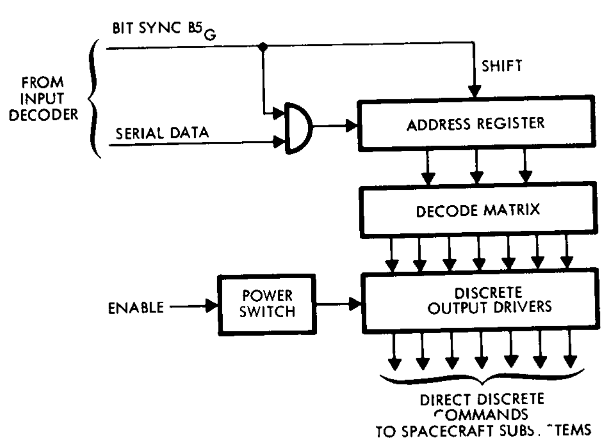


Figure 3-10

A BLOCK DIAGRAM OF OUTPUT DECODER is shown. The command data bits and bit sync pulses from the input decoder load and shift the address register. The discrete pulse output occurs when an enable signal is received from the input decoder.

load and shift the address register. The discrete pulse output occurs when an enable signal is received from the input decoder.

The contents of the address register are supplied to telemetry for ground verification of the received command. The register is sampled as part of the engineering data.

Table 3-2. Command Requirements*

Subsystem	Function	No. of Discrete Commands	No. of Serials	Remarks
Guidance and Control	Inertial hold mode	2		all axis inertial
	Enable sun acquisition	1		
	Canopus sensor on/off	2		
	Canopus sensor select	2		redundant unit selection
	TVC actuator selector	2		
	Gyro power on/off	2		
	Enable/disable accelerometer	2		
	TVC low/high gain select	2		
	TVC on/off	2		
	TVC test	1		
	Fine and coarse control select	2		deadband
	High and low thrust reaction control select	2		
	Roll spin-Canopus search	2		
	Star acquisition gate override	2		
	Roll incremental maneuver	2		2 deg steps
	Roll axis inhibit	2		
	High-gain antenna-hinge axis		1	12 bits
	High-gain antenna-shaft axis		1	12 bits
	Medium-gain antenna-hinge axis		1	12 bits
	Enable terminator-limb crossing detectors	2		
	Roll axis override	2		
	Enable maneuver	2		
	Canopus sensor update	1		
	Acquisition	2		
	Gyro reference assembly test	1		
	Gyro reference assembly select	2		
	Fine sun sensor select	2		
	Accelerometer select	2		
	Accelerometer test	1		

*This Command List gives all of the command functions performed by the Command Subsystem Decoder. The commands are grouped by their respective destination.

Table 3-2. Command Requirements* (Continued)

Subsystem	Function	No. of Discrete Commands	No. of Serials	Remarks
Capsule	Start/reset capsule S & T	2		
	Capsule power on/off	2		
	Open canister	1		
	Switch to capsule internal/ external power	2		
	Input to capsule register		6	
	Release capsule	1		
	Capsule thermal control on	1		
Power	Battery No. 1 charge control override on/off	2		
	Battery No. 2 charge control override on/off	2		
	Battery No. 3 charge control override on/off	2		
	Battery No. 1 disconnect open/close	2		
	Battery No. 2 disconnect open/close	2		
	Battery No. 3 disconnect open/close	2		
	Reset synchronization switch logic main/standby	2		
	Spacecraft power to capsule override on/off	2		
	Heater power override on/off	2		
	400 Hz inverter main/standby	2		
	Recondition battery No. 1 on/off	2		
	Recondition battery No. 2 on/off	2		
	Recondition battery No. 3 on/off	2		
	Non-essential load override on/off	2		
	Switch main/standby boost regulator	2		
Computer and Sequencer	Back-up C and S on/off	2		
	Serial data-memory store and register load		unlimited	33 bits required

*This Command List gives all of the command functions performed by the Command Subsystem Decoder. The commands are grouped by their respective destination.

Table 3-2. Command Requirements* (Continued)

Subsystem	Function	No. of Discrete Commands	No. of Serials	Remarks
Communi-cations	Transmitter/antenna	6		
	Exciter-power amplifier No. 1 on/off	2		
	Exciter-power amplifier No. 2 on/off	2		
	Low-power transmitter on/off	2		
	Receiver maximum gain-high-gain antenna	1		
	Receiver maximum gain-medium-gain antenna	1		
	Receiver maximum coverage	1		Special mode in event of loss of attitude control
	Non-coherent override on/off	2		
	Range code on/off	2		
	Low data rate capsule relay receiver on/off	2		
	High data rate capsule relay receiver on/off	2		
	Capsule relay link power on/off	2		
Telem-etry and Data Storage	Mode 1, maneuver engineering data	1		
	Mode 2, cruise engineering data	1		
	Mode 3, playback data	1		
	Mode 4, pre-maneuver data	1		
	Mode 5, photo-imaging data	1		
	Mode 6, capsule descent real-time data	1		
	51.2 kbits/sec bit rate	1		
	25.6 kbits/sec bit rate	1		
	12.8 kbits/sec bit rate	1		
	6.4 kbits/sec bit rate	1		failure transmission
	3.2 kbits/sec bit rate	1		failure transmission
	8 bits/sec bit rate	1		Emergency transmitter
	Store high rate capsule data	1		
	Store maneuver data	1		
	Biorthogonal coding on/off	2		
	Switch main/standby unit	2		
	Playback start/stop	12		C and S backup
	Record start/stop	12		C and S backup
	Record speed A/B	2		

*This Command List gives all of the command functions performed by the Command Subsystem Decoder. The commands are grouped by their respective destination.

Table 3-2. Command Requirements* (Continued)

Subsystem	Function	No. of Discrete Commands	No. of Serials	Remarks
Electrical Distri-bution	Open helium pressure valve	1		Orbit insertion
	Open fuel and oxidizer tank pressure valve	1		Orbit insertion
	Close pressure valves	1		
	Depressurize tanks	1		
	Release low-gain antenna	1		
	Release medium-gain antenna	1		
	Release high-gain antenna	1		
	Release PSP	2		2 positions
	Uncage gimbal PSP	1		
	Ordnance safe	1		
	Arm propulsion, first midcourse maneuver	1		
	Arm propulsion orbit insertion maneuver	1		
	Arm propulsion depressurization	1		
	Arm antenna release	1		
	Arm PSP release and uncaging	1		
	High-gain antenna hinge motor thermal control on/off	2		
	High-gain antenna shaft motor thermal control on/off	2		
	Medium-gain antenna hinge motor drive thermal control on/off	2		
	Close pressure valves	1		
	Open fuel and oxidize start tank solenoid valves	1		
	Fuel and oxidize motor driven pre-valves open/close	2		
	Engine solenoid valves open/close	2		Low thrust
	Engine solenoid valves open/close	2		High thrust
	Open helium pressure valve	1		Midcourse maneuver
	Open fuel and oxidize tank pressure valve	1		Midcourse maneuver

*This Command List gives all of the command functions performed by the Command Subsystem Decoder. The commands are grouped by their respective destination.

4. CAPSULE LINK SUBSYSTEM

4.1 SUMMARY

This section outlines those portions of the capsule radio link which form an integrated part of the spacecraft equipment and are retained on the spacecraft after capsule separation. As the complete capsule radio link has yet to be defined, the purpose of this section is to establish requirements placed on the spacecraft to support the capsule radio link.

The subsystem elements of the capsule radio link which mate directly with the spacecraft are:

- UHF receivers (2)
- Demodulators (2)
- Preamplifier
- Antenna.

Figure 4-1 is a functional block diagram of these elements showing interfaces with the spacecraft.

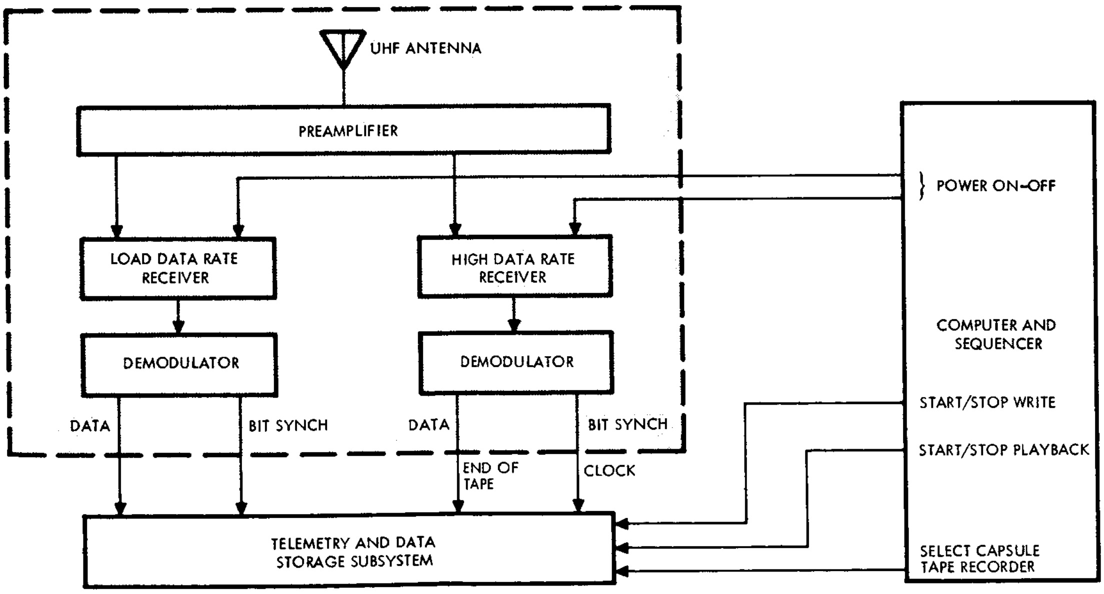


Figure 4-1
THE CAPSULE LINK includes the preamplifier, receivers, demodulators, and interface units in the telemetry and data storage subsystem and computer and sequencer.

4.2 SUBSYSTEM DESIGN AND PERFORMANCE

We presently assume that the capsule relay link operates with two separate data channels from separate capsule transmitters. A low data rate of 500 bits/sec comprises one data channel and operates from spacecraft-capsule separation to Mars touchdown. A high data rate channel of 100,000 bits/sec comprises the other data channel and operates from atmospheric entry to Mars touchdown (5 to 15 minutes, depending on entry flight time).

The capsule relay link must be designed to accommodate the varying communication distance. There are various types of descent trajectories which may be grouped into two classes, as shown in Figure 4-2. The first class of descent orbits requires separation at a true anomaly (measured clockwise from periapsis) of approximately 120 to 140 degrees of the spacecraft reference orbit and yields descent times not exceeding 2 hours. For this type of descent orbit the range is always less than 3000 kilometers. The second class of descent orbits requires separation at a true anomaly of approximately 240 to 270 degrees of the spacecraft reference orbit and yields descent times up to, but not exceeding, 12 hours. During the early portion of the descent in a Class II descent orbit, the separation range

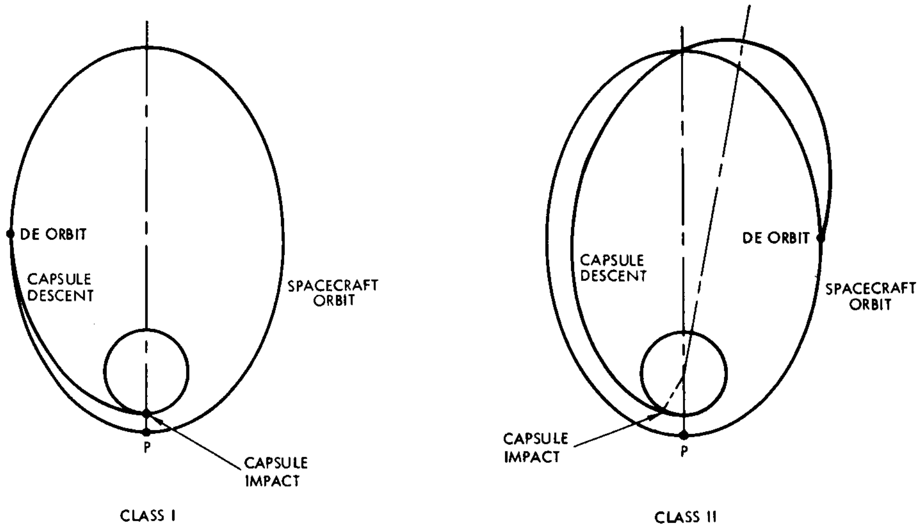


Figure 4-2
THE TWO TYPES OF CAPSULE DESCENT ORBITS, shown here schematically, result in communication distances between the spacecraft and the capsule of 3000 to 5000 km.

between flight spacecraft and capsule increases to approximately 5000 kilometers, but decreases to less than 3000 kilometers at the time of capsule atmospheric entry.

Until post-landing operations are defined, all data transmission and reception over the capsule radio link is assumed to take place during the descent time period.

All sequential events for capsule radio link operations are automatically programmed through the spacecraft computer and sequencer prior to separation. Those portions of the capsule link retained on the spacecraft subsequent to capsule separation derive controls for programmed sequences from the spacecraft computer and sequencer. A power on-off command and a separation signal are provided to control the operation of the capsule radio equipment and to switch the operation from hard line to radio link. Start-stop commands are supplied by the computer and sequencer to the tape recorder in the centralized telemetry and data handling unit for both write and playback of the capsule descent data.

The capsule data bit stream and bit sync are provided for both the low and high data rate links. Selected telemetry monitor points are provided for signal strength, regulated voltages, and in-lock indication. The two data rates arrive on different carriers. However, both the low and high data rate carriers are received by a single antenna, at a frequency in the 400 ±1 MHz band.

The low data rate carrier will carry telemetry at 500 bits/sec from separation to touchdown. This data will be relayed to earth in real-time by the spacecraft telemetry.

The high data rate carrier will carry television data at 100 kilobits/sec during the terminal descent phase of the mission. This data is recorded in the spacecraft and played back later over the S-band link.

4.3 COMPONENT DESCRIPTION

UHF Antenna

The UHF antenna occupies a volume of 27 x 27 x 10 inches and weighs 14 pounds. It is mounted on the aft side of the solar array at a clock angle of 230 degrees. The beam is pointed in the direction of the capsule descent trajectory at a nominal cone angle of 140 degrees and a nominal clock angle of 255 degrees. The antenna is designed to withstand a temperature range of -250 to +300°F, which is adequate for any spacecraft-sun angle.

The antenna, shown in Figure 4-3, is a quad-spiral array that provides a right-hand circularly polarized single-lobe radiation pattern, symmetrical about the axis of the array. The array has a gain of 10 decibels and a half-power beamwidth of 50 degrees. Each element in the array is a cavity-backed, two-arm Archimedian spiral fed by a balun transformer incorporated into the feed transmission line. Three integrated hybrid rings divide the power equally and maintain equal phase. Output connectors of the power divider are connected directly to the balun and transmission line of the element. The element design is selected for midband operation at the link frequency for maximum efficiency. The optimum spacing between elements of the array is near one-half wavelength center-to-center. The form factor of the antena allows it to be placed near the edge of the spacecraft, where a minimum of pattern interference from surrounding structure is experienced (Figure 4-4).

The use of two Canopus sensors 180 degrees apart allows the use of a single position of the UHF antenna for typical orbits designed for

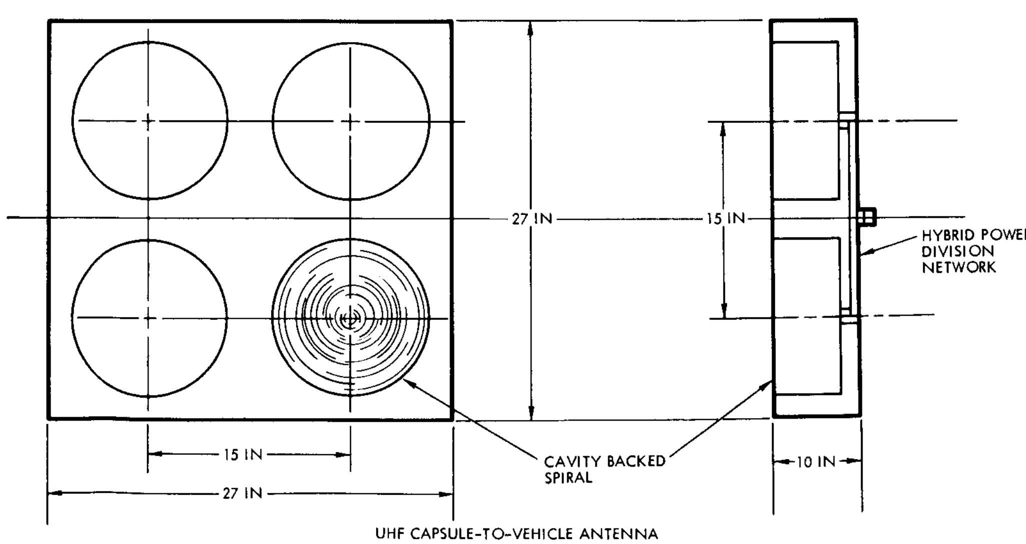


Figure 4-3
THE UHF ANTENNA is a quad-spiral array with a right-hand circularly polarized single lobe radiation pattern. It has 10 db gain and is 50 degrees wide at half-power.

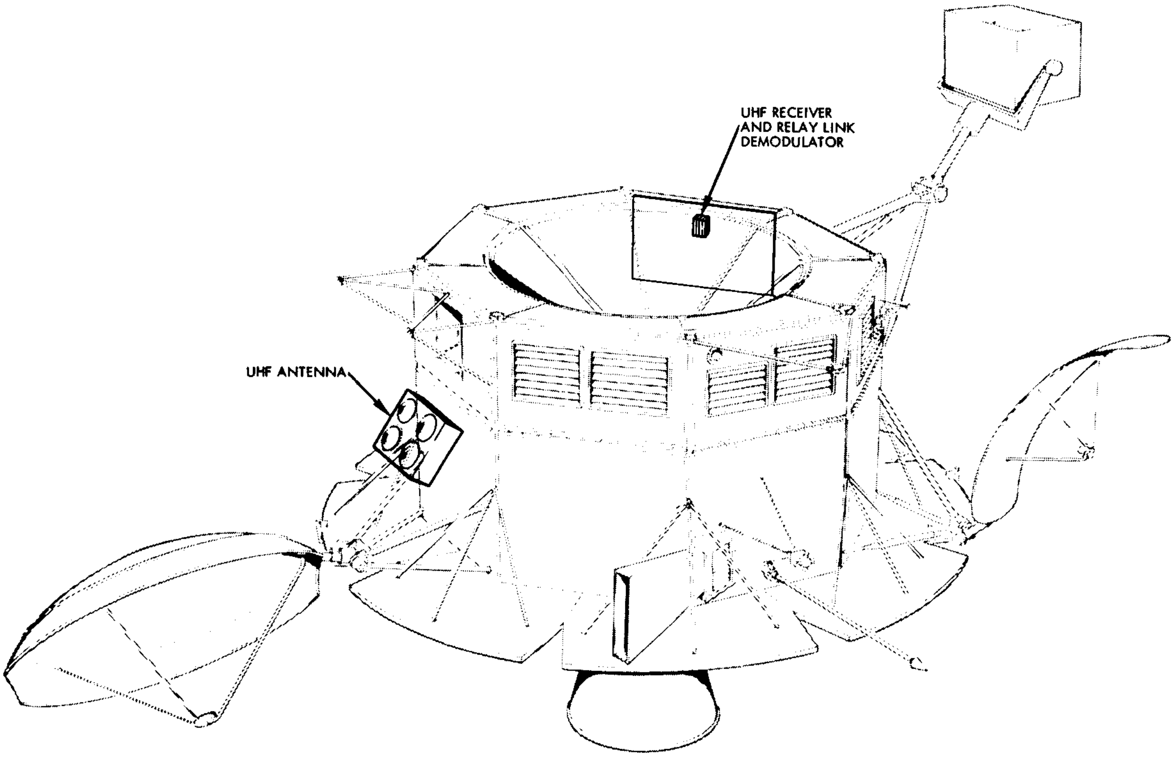


Figure 4-4
THE UHF ANTENNA FORM FACTOR enables it to be placed near the edge of the spacecraft, where a minimum of pattern interference from the surrounding structure is experienced.

Northern or Southern hemisphere coverage and capsule landing site location. Custom positioning of the antenna may be required if large orbit rotations are planned.

5. GUIDANCE AND CONTROL SUBSYSTEM

5.1 SUMMARY

The guidance and control subsystem provides three-axis attitude control of the planetary vehicle and spacecraft at all times following separation from the launch vehicle. It also measures vehicle acceleration during operation of the propulsion system and provides signals indicating the crossing of the Mars limb and terminator for use in the sequencing of science instruments. A preliminary specification for this subsystem appears in Figure 5-1.

In the design of the guidance and control subsystem the aim has been to achieve the maximum reliability with a performance that meets mission requirements. A proven subsystem design with proven components is recommended. The system can operate in various modes to provide backup protection if a component fails. Changes in the system required for the upgraded spacecraft are minimal, consisting chiefly of provision of a larger gas supply in the reaction control system to accommodate the larger vehicle inertias, tighter limit cycles, and larger solar pressure torques.

Immediately following separation of the planetary vehicle from the S IV-B launch vehicle, the guidance and control subsystem reduces tumbling rates imparted by the separation to a very low level (approximately 0.01 deg/sec) which subsequently enables a good sun acquisition response. The vehicle operates in this low angular rate during the period of sun eclipse following separation.

Upon emergence into the sunlight, the celestial reference acquisition process is begun. The pitch and yaw axes are stabilized with respect to the sun, thereby maintaining the solar array surface normal to the sunlight as well as providing a celestial reference for trajectory correction maneuvers. The roll axis is stabilized with respect to the star Canopus. This orientation is maintained throughout interplanetary cruise, except for periods when trajectory correction maneuvers are made. For the cruise modes of operation a nitrogen gas reaction control system with heated thrusters is used to provide control torques.

PRELIMINARY SPECIFICATION

Guidance and Control Subsystem

Purpose

The guidance and control subsystem provides continuous three axis attitude control of the planetary vehicle using either a sun-canopus celestial reference system or an on-board inertial reference system. Capability is provided for reorientation in all three axis and for the measurement of incremental velocity in the thrust axis. An indication of the crossing of the Mars dawn and evening terminators is provided.

Performance

ATTITUDE CONTROL

Post Separation Eclipse

Reduces tipoff angular rates of 1.5 deg/sec max to 0.01 deg/sec and maintains throughout the period of eclipse.

Celestial Reference Acquisition

Automatically acquires the sun and Canopus from arbitrary attitude.

Earth-Mars Cruise

Maintains pointing accuracy of ± 0.6 deg, 3σ each axis with respect to celestial reference frame.

Trajectory Correction

Orients to desired direction, provides thrust vector pointing accuracy of ± 0.43 deg, 3σ each axis with respect to the celestial reference frame.

Orbit Insertion

Orients to desired direction, provides thrust vector pointing accuracy of ± 0.43 deg, 3σ each axis with respect to the celestial reference frame.

Mars Orbit

Same as Earth-Mars Cruise except provides for three axis inertial operation during periods of celestial reference occulation.

Mars Orbit Trim

Orients to desired direction, provides thrust vector pointing accuracy of ± 0.43 deg, 3σ each axis with respect to the celestial reference frame.

Capsule Separation

Orients to desired direction, maintains pointing accuracy of 0.73 deg, 3σ each axis.

Photo-Imaging

Maintains spacecraft pointing accuracy of ± 0.35 deg, 3σ each axis and angular rates less than 10 deg/hr.

Intertial Hold

Provides for attitude control using on-board inertial reference with attitude drift less than 0.5 deg/hr.

VELOCITY MEASUREMENT PERFORMANCE

Measures velocity increments of up to 3000 meters/sec with an accuracy of 0.1 percent and with a resolution of 0.005 meters/sec.

TERMINATOR CROSSING INDICATION PERFORMANCE

Provides an indication of the passage of the sunlit portion of the Mars disc through a cone angle of 90 degrees in the spacecraft coordinate system.

Physical Characteristics

Weight:	279 lb
Power:	2051 watts (average during maneuvers)

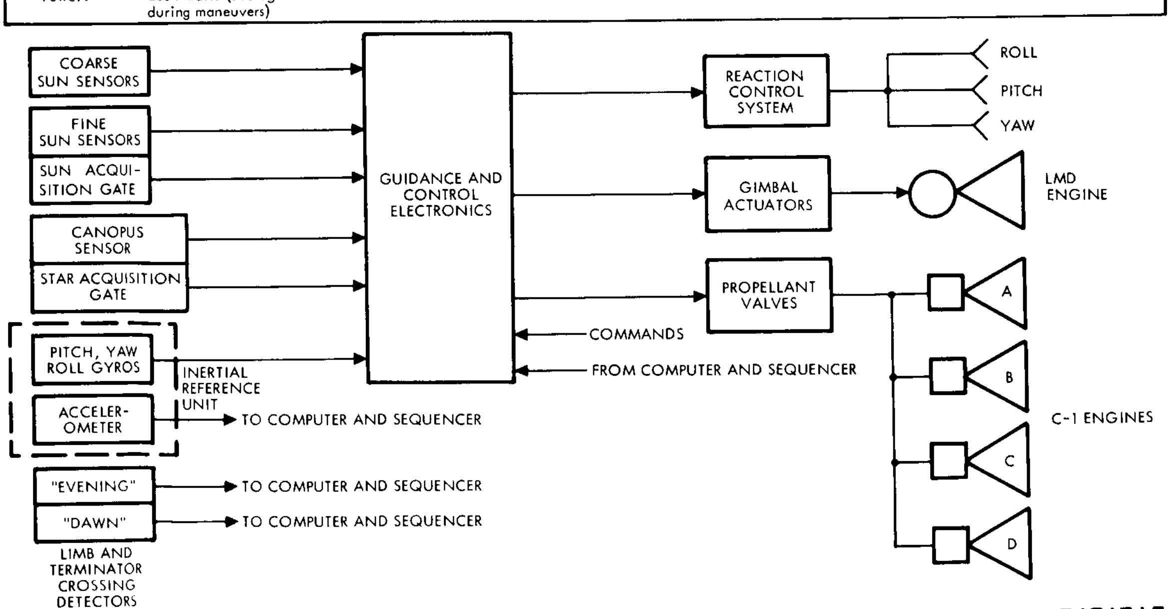


Figure 5-1

For the powered maneuvers (i.e., trajectory correction, orbit insertion, and orbital trim) and for capsule separation the guidance and control subsystem reorients the spacecraft to the required attitude. This is accomplished in accordance with ground commands that were previously stored in the computer and sequencer and are then issued to the guidance and control subsystem. The attitude reference for these operations is an inertial reference unit, the elements of which are initially aligned with the celestial reference frame. After each maneuver the cruise mode of operation is re-established with re-acquisition of the celestial references.

During operations in Mars orbit the guidance and control subsystem maintains three-axis stabilization of the vehicle using again the sun-Canopus reference. It also provides signals indicating crossing of the Mars limb terminator. In the Mars orbit, prior to the predetermined periods of sun or Canopus occultation, the inertial reference unit is energized and control is automatically transferred to the inertial references system until the celestial references are again visible.

Thrust vector control of the LMD engine is provided by gimballing the engine with respect to the pitch and yaw axes under the control of the inertial reference unit.

A simplified block diagram of the guidance and control subsystem showing the arrangement of major components is shown in Figure 5-1. There is full redundancy of all critical components to prevent loss of the mission in case of a component failure. Table 5-1 lists the units or assemblies comprising the subsystem and indicates their weight and the use of redundancy. The location and arrangement of these components on the spacecraft equipment module is shown in Figure 5-2.

Figure 5-3 illustrates the interface relationships of the guidance and control subsystems and other principal subsystems of the Voyager vehicle.

5.2 SUBSYSTEM DESIGN AND PERFORMANCE

5.2.1 Configuration Selection

A fully attitude controlled configuration has been selected for the Voyager spacecraft. This approach is justified by the requirements of interfacing spacecraft subsystems and the requirements for accurate

Table 5-1. Guidance and Control Subsystem Equipment List Indicating Equipment Weight and Use of Redundancy for Reliable Operation

Item No.	Description	No. Required Spacecraft	Unit Weight (lb)	Redundancy
1	Inertial Reference Assembly (3 gyros + 1 accelerometer)	2	25	Redundant units
2	Canopus Sensor	2	7	Redundant units mounted 180 clock degrees apart
3	Coarse Sun Sensor	4	3	Operational redundancy
4	Fine Sun Sensor	2	0.2	Redundant units
5	Guidance and Control Electronics Assembly	1	13	Internal circuitry and components redundant
6	TVC Actuators	2 (1 pitch, 1 yaw)	24	Internal components redundant
7	Reaction Control Subsystem (tanks, valves, regulators, plumbing, etc.)	1	158 including (61 lb of gas)	2 redundant "1/2" systems
8	Limb and Terminator Detectors	2	0.6	Operational redundancy

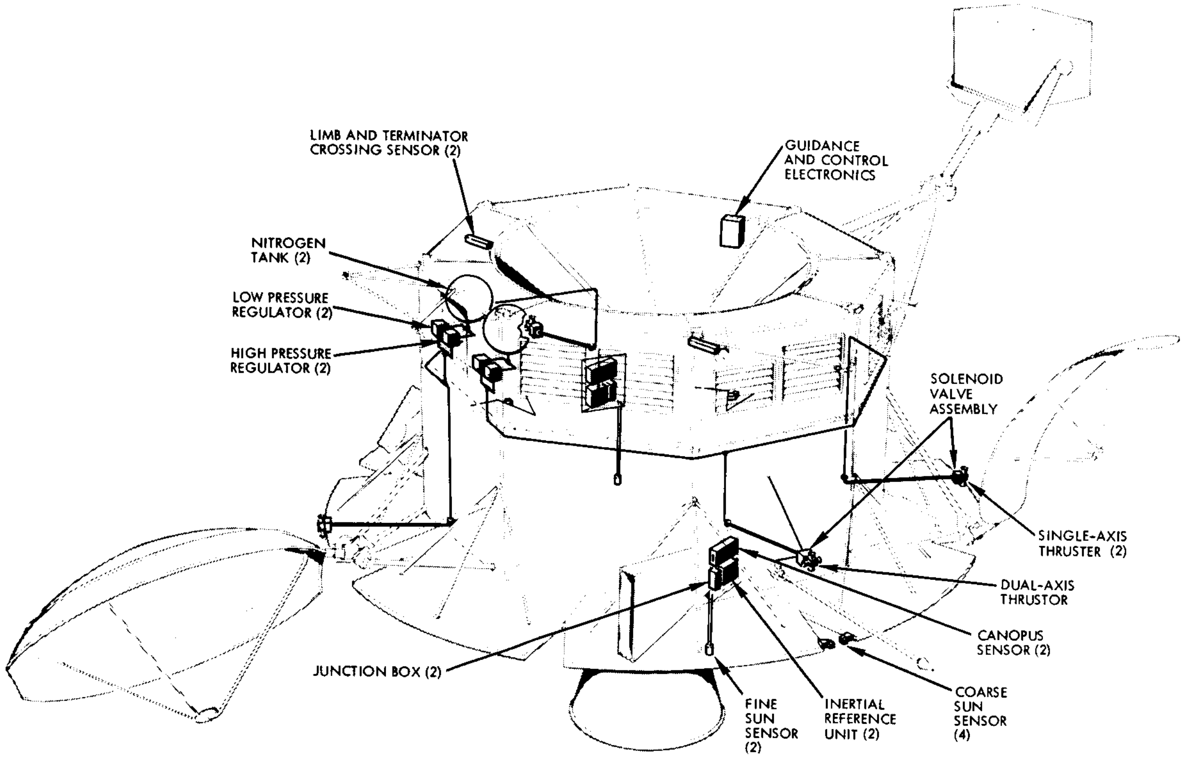


Figure 5-2

THE GUIDANCE AND CONTROL SUBSYSTEM includes redundant fine sun sensors, canopus sensors, and interial reference units. The reaction control system makes use of electrically heated nitrogen gas during cruise.

direction of the thrust vector generated by operation of the main engine or C-1 engines. All of the requirements leading to the selection of an attitude control configuration can be summarized as follows:

- The spacecraft must provide a stable platform oriented in a known direction for directional pointing of the gimballed antennas and the Planetary Scan Platform.

- The solar panels must be oriented in a direction such that they are approximately normal to the sun line.

- The thrust vector generated by the main engine or C-1 engines during midcourse maneuvers, orbital insertion and orbit trim maneuvers must be accurately directed and measured.

- The spacecraft must operate with a stable, controlled attitude during main or C-1 engine operation to avoid excessive structural loading.

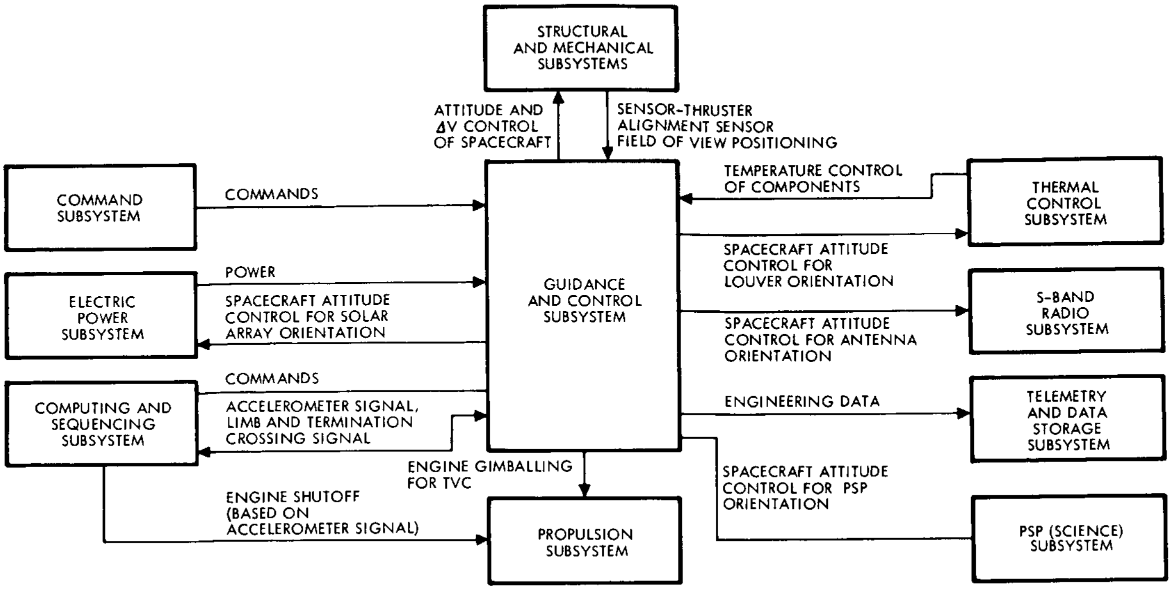


Figure 5-3
INTERFACES TO THE GUIDANCE AND CONTROL SUBSYSTEM are many and varied. Initiation commands come from the command subsystem or the computer and sequencer subsystem. Prime output is the control of spacecraft attitude for the proper functioning of other subsystems.

- The spacecraft attitude must be directed and held in a specified position for operation of the capsule separation sequence.

5.2.1.1 Celestial Sensors

A sun-Canopus based reference system has been chosen as the simplest system to meet the basic requirements of the Voyager mission. The use of a control law, which provides for one of the axes of the vehicle to point toward the sun at all times (except during maneuvers), also simplifies the solar panel design in that the solar panels can now be permanently fixed to the vehicle structure with the proper alignment with respect to sunlight. The use of the sun as one of the celestial references is appropriate in view of the relative operational simplicity of solar sensors as opposed to sensors operating on radiation from other celestial bodies. The selection of the second celestial reference as required for full three-axis control is more difficult inasmuch as there is no absolutely superior choice. Several alternatives such as Canopus, Mars and other bright stars are available and their use has been studied. All things considered, however, the use of Canopus appears to be superior from the standpoint of the geometrical relationships and the availability of design techniques and background experience for Canopus-sensing hardware.

For the sun sensors a simple coarse-fine arrangement has been selected using silicon photovoltaic cells. The coarse sensors provide a full 4π steradian field of view enabling a fully automatic sun acquisition sequence. The fine sun sensor consists of a quad cell arranged with a shade to provide a relatively narrow field of view. A switching sensor provides a signal to automatically transfer control from the coarse sensor to the fine sensor or vice versa.

5.2.1.2 Inertial Reference Sensors

Attitude reference requirements imposed by reorientation for mid-course maneuvers, orbit insertion, capsule separation, and orbit trim maneuvers are met in the recommended design by the use of gyroscopes. It is possible to use sun aspect sensors for this application; however, gyros offer flexibility and the use of well known techniques and components. Accordingly, the recommended design employs an inertial reference unit containing three single-degree-of-freedom floated rate-integrating gyroscopes which can be operated as either position or angular rate sensors. One gyro is located on each control axis, and by maneuvering one axis at a time, satisfactory reorientation accuracy can be obtained with a very simple system mechanization.

During operation of the main engine or C-1 engines the same gyros are used as vehicle attitude sensors in an autopilot-type mechanization, thus providing the required thrust vector directional control accuracy. A single linear accelerometer also included in the inertial reference unit provides an indication of velocity gained along the thrust axis.

5.2.1.3 Control Torque Sources

Control torques for periods other than during velocity correction can be generated by mass expulsion devices, reaction wheels, or solar vanes. Mass expulsion devices, especially gas thrusters, have received extensive application and are well understood. Consequently, a cold gas system with heated thrusters has been chosen as the reaction control system for the recommended design. To minimize the amount of control propellant required, the gas is heated, only during the cruise mode by means of an electric heater to $1500^{\circ}F$ at the nozzle, thus effectively doubling the available specific impulse.

Since there are no significant cyclical disturbance torques, reaction wheels offer no advantages for the Voyager mission. Solar vane systems appear to be best suited as trim devices because of their low frequency response. As trim devices, the solar vanes might be used in parallel with the cold gas reaction control system. However, they have not been included in the recommended design because the resulting saving in gas is not significant enough to justify the added complexity and weight.

Alternate reaction control systems using propellants heated by combustion or decomposition have been investigated but discarded in favor of the cold gas electrically heated system because of the simplicity and proven reliability of the latter system.

Control torques during periods of operation of the main engine can be obtained by gimballing the engine or from the reaction control system. Analyses have indicated that the gas requirement for the production of adequate control torque in the pitch and yaw axes during main engine operation is prohibitively large due to vehicle center-of-gravity offsets and engine and thrust misalignments. Accordingly, for the recommended design, the approach of gimballing the main engine by means of electric gimbal actuators has been chosen to provide the pitch and yaw control torques during main engine operation. The Lunar Module engine is well suited to this application and presents no problems or need for the development of new techniques. Since the disturbance torques about the roll axis are small, consisting primarily of propellant swirl torques, control in this axis can be achieved with the gas system without excessive gas usage.

5.2.1.4 Control Modes

The Voyager mission requires that the guidance and control subsystem provide several distinct modes of operation to suit the requirements of the various mission phases. These control modes make use of various combinations of sensors, control torque sources, and electronics which mechanize a particular control law. The selection of the various sensors, actuators, and electronics is determined by the switching logic incorporated in the subsystem in response to external commands or discrete signals generated within the subsystem itself. Table 5-2 illustrates the utilization of sensors and other components for the various modes of operation.

Table 5-2. Utilization of Sensors, Reaction Control System and Actuators for the Various Operating Modes of the Guidance and Control Subsystem. Also Illustrates the Subsystem Performance in Terms of Limit Cycle Amplitudes and Body Angular Rates for Each of the Operating Modes

	Sensors Usage								Subsystem Operation						Rate Damping Method		
				Gyro		RCS Thrust Level			Limit Cycle Amp		Turn Rates						
Operating	Coarse Sun	Fine Sun	Canopus	Rate	Pos	3 lb	0.2 lb	Actuator Usage	$\pm 0.5^{\circ}$	$\pm 0.25^{\circ}$	0.2°/sec	0.1°/sec	0.01°/sec	$<10^{\circ}$/hr	Gyro Rate	DRI	Lead/Lag
Test	X	X	X	X	X			X									
Boost				X													
Rate null				X		X							X		X		
Sun Acquisition	X	X		X		X					X				X		
Canopus Acquisition		X	X	X		X						X			X		
Cruise		X	X				X		X					X		X	
Manuever					X	X					X						X
Inertial Hold					X	X			X					X		X	
Actuator Test		X	X			X		X (test only)	X					X		X	
Thrust Vector Control					X	X (roll only)			(roll only)								X
Photo Imaging		X	X		*		X			X				X		X	

*During sun or Canopus occultation only

Of considerable importance from a performance standpoint is the selection of the proper mechanization for a particular operating mode. For example, since the greatest percentage of the Voyager mission time for both the transit and orbital phases is spent in the cruise mode, it is essential that this mode combine the maximum possible reliability of operation with minimum control gas consumption. Accordingly, detailed analyses have been conducted to determine the optimum thrust levels, control deadzones, and method of providing damping. While these analyses are discussed in detail later in this section, the mechanization selected for the guidance and control system can briefly be summarized as follows.

For the cruise mode, a form of derived-rate increment modulator control which consists of a low pass filter feedback loop around a fixed level on-off controller has been selected. To reduce gas consumption, a reaction control system thrust level of 0.2 pounds is used during the cruise mode.

In order to improve the system response during acquisition and maneuver, the thrust level is raised to 3 pounds by increasing the system pressure. In addition, a lead-lag switching circuit in parallel with the DRI controller is included for response improvement. During sun acquisition, the rate gyro output signals are summed with the coarse sun sensor output signals to produce a constant rate of turn. During Canopus acquisition, the rate gyro output signals are summed with rate command voltages, thereby producing a constant roll rate for Canopus search.

Attitude control during periods of operation of the main engine is achieved by means of a conventional autopilot-type loop using the three gyros operating in the position mode. The required rate damping is obtained by means of lead-lag networks operating on the gyro output signals. To reduce thrust-vector directional errors due to center of mass displacement and thrust misalignment, the gyro output is integrated and added to the control signal. A fixed bias, required to overcome the effect of the predictable center of mass offset, is also inserted into the loop. It is also possible to make use of more sophisticated techniques for the correction of errors due to these effects as for instance, the use of engine gimbal angle feedback or the use of control signals generated by two lateral accelerometers; but the Voyager mission can be accomplished with the simpler, more reliable, mechanization described.

5.2.1.5 Backup Operating Modes

A form of backup operation is provided for all modes to enhance the overall probability of mission success. For instance, the gyros form a backup to the celestial sensors for the cruise mode in that the inertial hold mode can be utilized during periods of solar eclipse or when occultation or stray light prevents proper operation of the Canopus sensor. Likewise, provision has been made for a ground-controlled incremental roll maneuver similar to the roll axis maneuver mode with the exception that the precision torquing current is controlled by fixed timers which results in a fixed incremental attitude change. In addition, provision is made for direct control of the roll axis by ground-initiated commands to aid in Canopus acquisition and high-gain antenna pointing.

All of the critical components of the guidance and control subsystem are backed up by redundancy. Two complete inertial reference units as well as two Canopus sensors and two fine sun sensors are carried on board to provide for failure of any one of these components. The two Canopus sensors are located 180 clock angle degrees apart on the vehicle so that one can be used for north-inclined Martian orbits and the other for south inclined orbits. In addition to providing for failure protection this scheme enables the optimization of vehicle attitude for photo-imaging and allows the use of a single capsule relay link antenna in the two different types of orbits.

Certain components such as the electronics assembly and the TVC actuators are internally redundant in that their circuitry and major components are arranged such that failures are automatically detected and compensated. In the electronics assembly this is accomplished by the extensive use of triply redundant voting-type circuits.

The reaction control system makes use of the "half" system concept to provide protection in case of failure of one of its components. With this approach three times the mission-required gas is carried in two completely independent systems which allows the complete failure (loss of gas) of one system while still retaining the capability for completion of the mission.

During engine operation, if one of the accelerometers fails, the on-board failure detection circuit will automatically switch to the other standby accelerometer. In case both accelerometers or their associated electronics circuits fail to produce an engine shut-down signal, a timed signal, stored prior to engine firing in the computer and sequencer subsystem with a time increment slightly larger than the commanded engine firing time, serves as a backup command to shut down the engine.

5.2.2 Subsystem Operation

The functional operation of the guidance and control subsystem is best described by reference to the various modes of operation in approximately the same order in which they would be used in the Voyager mission. Table 5-3 shows in tabular form the various modes of operation and the utilization of the sensors and torque sources (reaction control system and actuators).

5.2.2.1 Test Mode

This mode of operation is included primarily for prelaunch checkout of the subsystem. By its use all operating channels of the subsystem can be remotely stimulated by means of electrical signals simulating the outputs of the various sensors. The outputs of the RCS valves, actuators, and C-1 engine valves are then observed via hardline or telemetry and checked for polarity and correct amplitude of function. Using this mode the gyros and accelerometers in both the primary and secondary inertial reference assemblies can be checked for operational readiness by applying appropriate torquing currents and observing their outputs.

5.2.2.2 Boost Mode

In this mode of operation, which is used during the boost phase of the mission up to and including the final burn of the S-IVB, the guidance and control subsystem is electrically energized but the reaction control system is disabled to prevent the unnecessary expenditure of gas. All

Table 5-3. Guidance and Control Subsystem*

Item	Subsystem Operating Mode	When Used	Subsystem Configuration and Operation
1.	Test	Prior to launch	Sensors are electrically stimulated and subsystem outputs are checked for correctness.
2.	Booster Operation	From launch to separation from SIVB	Subsystem is energized, sensors are disconnected and protected, RCS and actuators are disabled.
3.	Rate Nulling	Following separation and during post-separation eclipse	Gyros are operated in rate mode, subsystem operates to reduce rates to 0.01 deg/sec or less using 3-lb jets.
4.	Sun Acquisition	Following post-separation eclipse and as required for sun acquisition from arbitrary attitude	Course sun sensor signals enable orientation of the vehicle with solar panels normal to the sun. 3-lb jets are used and control is shifted to fine sun sensor at ±10 deg of error. Gyros are operated in the rate mode to provide damping.
5.	Canopus Acquisition (roll spin)	Following initial sun acquisition and as required for Canopus reacquisition	A roll rate of 0.1 deg/sec is established using the roll gyro in the rate mode. Canopus is acquired when it appears in the field of view of the Canopus sensor. 3-lb jets are used until acquisition is verified at which time RCS is switched to 0.2-lb thrust level.
6.	Cruise	During Earth-Mars transit and during Mars orbit except during periods of photo imaging or occultation of the celestial references	Attitude reference is provided by celestial sensors, rate damping is by means of DRI networks. RCS operates on a 0.2-lb thrust level providing ±0.5 deg attitude limit cycle.
7.	Maneuver	Prior to and following main or C-1 engine operation or as required for antenna or PSP positioning, capsule separation, or re-acquisition of the celestial references	Attitude reference is provided by gyros operated in position mode. These are torqued at 0.2 deg/sec in each axis in turn for time required to establish the desired attitude. At start of reorientation gyros are initiated from celestial sensors and RCS is switched from 0.2- to 3-lb thrust level. A gyro test is also performed as a part of the gyro initiation.
8.	Inertial Hold	During Mars orbit for occultation of the celestial references and as-required during maneuvers	Similar to reorientation mode except gyros are not torqued so that initial attitude is held fixed.
9.	Actuator Test	Prior to operation of the main engine or C-1 engines	Actuators are energized and stimulated with electrical test signal. Outputs are observed for correctness. Balance of the subsystem remains in the same configuration as for inertial hold.
10.	Thrust Vector Control	During operation of the main engine or C-1 engines for midcourse corrections, orbit injection or orbital trim	Attitude reference is provided by gyros operating in the position mode. RCS operates at the 3-lb level and pitch and yaw thrusters are disabled. Pitch and yaw control is obtained by gimballing main engine with actuators.
11.	Photo Imaging	During Mars orbit when photo imaging operation of the PSP is desired	Similar to cruise mode or inertial hold mode except that RCS is operated at the 0.2-lb level and the attitude limit cycle is reduced from ±0.5 deg to 0.25 deg.

*Operating Modes Table Indicating the Usage of Each of the Various Operating Modes and the Configuration of the Subsystem for Each of the Modes.

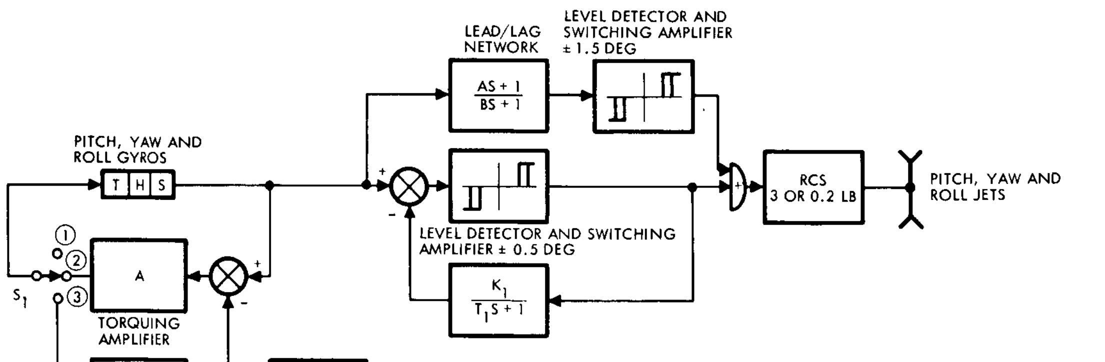
sensors and electronics are in operating condition and the gyros are at operating temperature with the spin motors running and the gimbals caged via the rate loop.

5.2.2.3 Rate Null Mode

The rate nulling mode as described in Figure 5-4 is used immediately following separation of the planetary vehicle from the S-IVB stage. The pitch, yaw, and roll gyros are operated in the rate mode and their output signals are used to reduce the spacecraft body angular rates from a maximum of 1.5 deg/sec dur to tipoff, in each axis, to less than 0.01 deg/sec. Using the RCS set at the 3-pound thrust level this operation is performed within 15 seconds and thereafter the control system operates to keep the rates below this level. The control system continues in this mode of operation throughout the period of eclipse following separation from the S-IVB.

5.2.2.4 Sun Acquisition Mode

Upon emergence of the spacecraft into sunlight the sun acquisition mode is automatically initiated by the illumination of the coarse sun sensors.

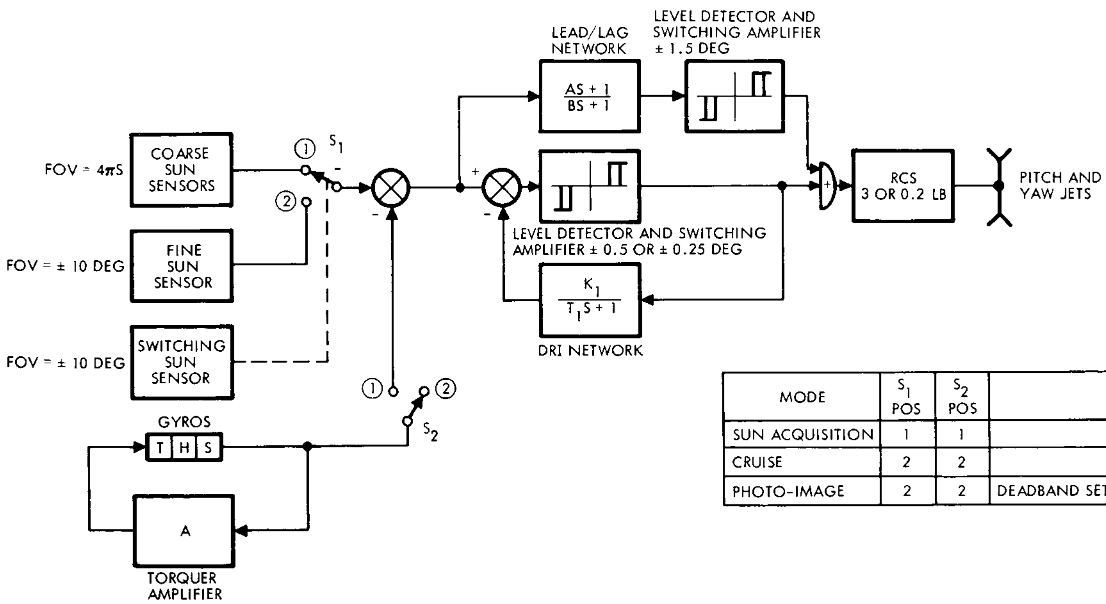


MODE	S_1 POS	S_2 POS	
RATE NULLING	2	2	
MANEUVER	3	1	S_1 INITIALLY SET TO 2
INERTIAL HOLD	1	2	

Figure 5-4

RATE NULLING, MANEUVER AND INERTIAL HOLD modes of operation of the guidance and control subsystem involve this arrangement of components. For these modes, switches are set according to the table.

The arrangement of components of the subsystem for this mode of operation is shown in Figure 5-5. This block diagram illustrates only the operation of the pitch and yaw channels, since the roll channel is disabled, by disconnecting the roll RCS valves during sun acquisition. At the beginning of sun acquisition the outputs of the coarse sun sensors are connected to the control switching amplifiers and both the lead/lag and the DRI channels are operated in parallel. The gyros are operated in the rate mode and their outputs are subtracted from the saturated sensor outputs to establish a rate command to the vehicle. Although the derived rate circuits are not required for this mode, they are retained to eliminate switching and to provide a fine pulsing augmentation of the lead-lag response. The output signals of the coarse sun sensors saturate at 10-degree error and since the rate gyro gain has been selected as 50 seconds, the saturated region from 10 to 170 degrees will command a spacecraft rate of 0.2 deg/sec. Illumination of the fine sun sensor occurs when the planetary vehicle is within 10 degrees of the sun line and results in activation of the switching sensor to switch the input from the coarse to the fine sun sensor.



MODE	S_1 POS	S_2 POS	
SUN ACQUISITION	1	1	
CRUISE	2	2	
PHOTO-IMAGE	2	2	DEADBAND SET TO ± 0.25 DEG

Figure 5-5

SUN ACQUISITION, CRUISE AND PHOTO-IMAGING MODES OF OPERATION OF THE GUIDANCE AND CONTROL SUBSYSTEM involve this arrangement of components in the pitch and yaw channels. For these modes, the switches shown are set according to the table

Because the output of the fine sensor decreases linearly from 10 degrees, the angular rate is commanded to decrease from 0.2 deg/sec to nominally zero as the pointing error is nulled. Upon completion of this mode the roll axis of the vehicle will be directed toward the sun and the vehicle will be operating in a stable limit cycle about the ±0.5 degree deadzone in the pitch and yaw axes.

5.2.2.5 Canopus Acquisition Mode

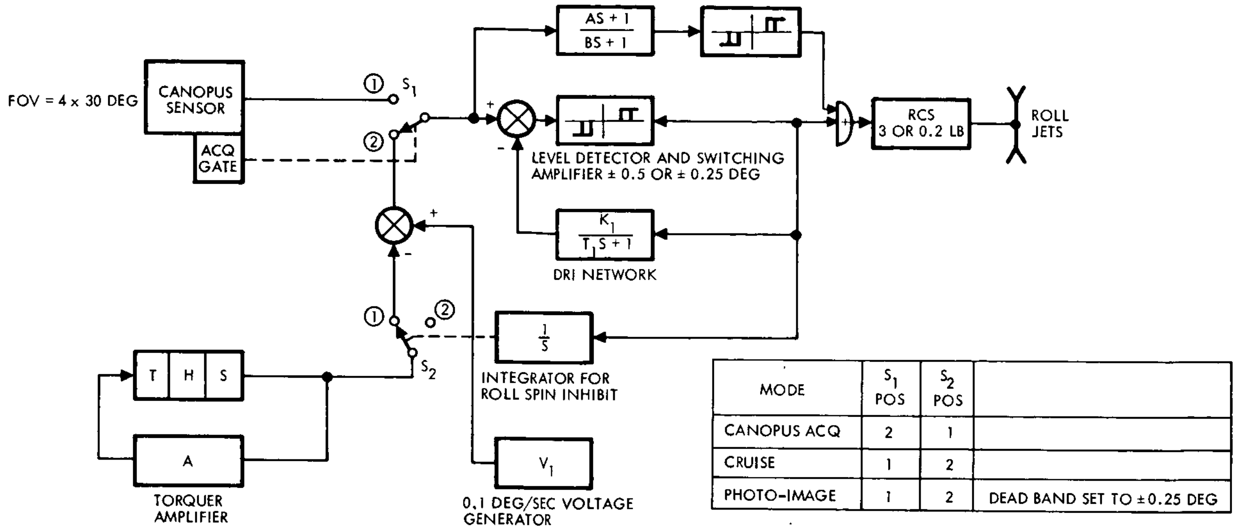
When the vehicle roll axis has acquired the sun as signalled by the appearance of the sun in the field of view of the sun acquisition sensor and a low value of error signal from the fine sun sensor, the Canopus acquisition mode is commanded by the computer and sequencer.

For this mode a controlled roll rate of 0.1 deg/sec is established with the arrangement of components shown in Figure 5-6. The Canopus sensor shutter is opened and its acquisition field of view of 4.0 degrees (in clock) by 30 degrees (in cone) is caused to sweep the star field. When a star within the chosen brightness range appears in the field of view, a star acquisition signal is generated which switches the control system from gyros to the output of the Canopus sensor.

The acquisition of Canopus is verified by prepositioning the high-gain antenna so that the antenna beam intersects the earth when the spacecraft is in the proper post-Canopus acquisition attitude. If a wrong star is acquired as indicated by the absence of the RF signal a command is sent to revert the control system to the Canopus acquisition mode and the roll search is resumed until Canopus is located. A further verification of Canopus is the pitch error signal from the star tracker which provides the sun-spacecraft-star angle.

5.2.2.6 Cruise Mode

With the completion of sun and Canopus acquisition, the control system is operating in the cruise mode. As shown in Figures 5-5 and 5-6, this mode provides pitch and yaw control derived from fine sun sensor signals and roll control derived from the Canopus sensor signals. For this mode the RCS is switched to the 0.2-pound thrust level, and since the



MODE	S_1 POS	S_2 POS	
CANOPUS ACQ	2	1	
CRUISE	1	2	
PHOTO-IMAGE	1	2	DEAD BAND SET TO ±0.25 DEG

Figure 5-6
CANOPUS ACQUISITION, CRUISE AND PHOTO IMAGING MODES OF OPERATION OF THE GUIDANCE AND CONTROL SUBSYSTEM involve this arrangement of components in the roll channel. For these modes,the switches shown are set according to the table.

gyros are turned off to conserve power, rate damping is provided by the derived rate increment networks. The vehicle itself operates in stable limit cycles within the ±0.5 degree deadband in each axis with limit cycle rates less than 10 deg/hr.

The cruise mode of operation is used for both transit and orbital phases of the mission. During the photo-imaging periods of the orbital phase the limit cycle deadband is reduced to ±0.25 degree to increase the pointing accuracy of the PSP.

5.2.2.7 Maneuver Mode

This mode is used before or after a midcourse trajectory correction, orbital insertion, capsule separation, or orbit trim operations. Essentially, it consists of a reorientation of the vehicle to the desired direction using the on-board inertial reference assembly. The arrangement of subsystem components for this mode is shown in Figure 5-4. Approximately one hour prior to a maneuver the computer and sequencer commands the inertial reference unit on to enable it to warm up and stabilize. After proper warmup, a gyro test is performed to verify proper performance of the gyros. At the start of the maneuver the RCS is switched to a 3-pound thrust level and the control system is switched from the celestial sensors to the gyros. The gyros which are operated in the position mode have been previously initialized so that they are compensated for the attitude error existing at the time of transfer. The gyros are then torqued at the rate of 0.2 deg/sec by means of precision current generators and the vehicle turns through a corresponding angle under the action of the control system. Accordingly commanded turns are performed in the pitch, yaw, and roll axes in turn in response to polarity and turning time data transmitted by radio command and stored in the computer and sequencer.

Following completion of a reorientation maneuver, the new attitude is maintained by continuous inertial control.

5.2.2.8 Inertial Hold Mode

The operation of the inertial hold mode is similar to the operation of the maneuver mode except that the gyros are not torqued. Therefore, the vehicle maintains a constant attitude in inertial space within the limitations of gyro drift and the vehicle operating limit cycle. This mode is used subsequent to the maneuver mode and at other times when it is desired to maintain a constant attitude without continuous reference to the celestial sensors, as for instance during sun and Canopus occultations in the orbital phase of the mission.

The inertial hold mode can be used indefinitely subject only to gyro drift. However, it is primarily intended for relatively short periods of operation (several hours). Under these conditions, the celestial references can be reacquired simply by repeating the preceding maneuver in reverse, that is torquing the gyros in the opposite direction at the same rate and for the same time under the command of the computer and sequencer. If it is not desirable to do this, or if the inertial hold mode has been used long enough that gyro drift renders this approach useless, the celestial references can be acquired automatically using the sun acquisition and Canopus acquisition modes.

5.2.2.9 Actuator Test Mode

This mode is used prior to operation of the main engine to assure proper operation of the TVC channels and the gimbal actuators. Essentially it involves simulating an error signal at the inputs of the TVC channels, energizing the actuators and observing their outputs for correct polarity and magnitude.

5.2.2.10 Thrust Vector Control Mode

Figure 5-7 shows the arrangement of subsystem components for this mode of operation, which is used during operation of the main engines or C-1 engines for midcourse trajectory corrections, orbit insertion, and orbit trim. For this mode the gyros are operated as position sensors and their outputs control the thrust vector direction via gimballing of the main engine or duty cycle modulation of the C-1 engines.

The gyros are aligned to the RCS axes and the engine actuators are aligned 40 deg from these axes. The pitch and yaw commands from the filtered gyro error signals are transformed with the fixed alignment angle to obtain the actuator commands.

Since the C-1 engines are aligned to the actuators, the actuator commands are conveniently employed for attitude control during C-1

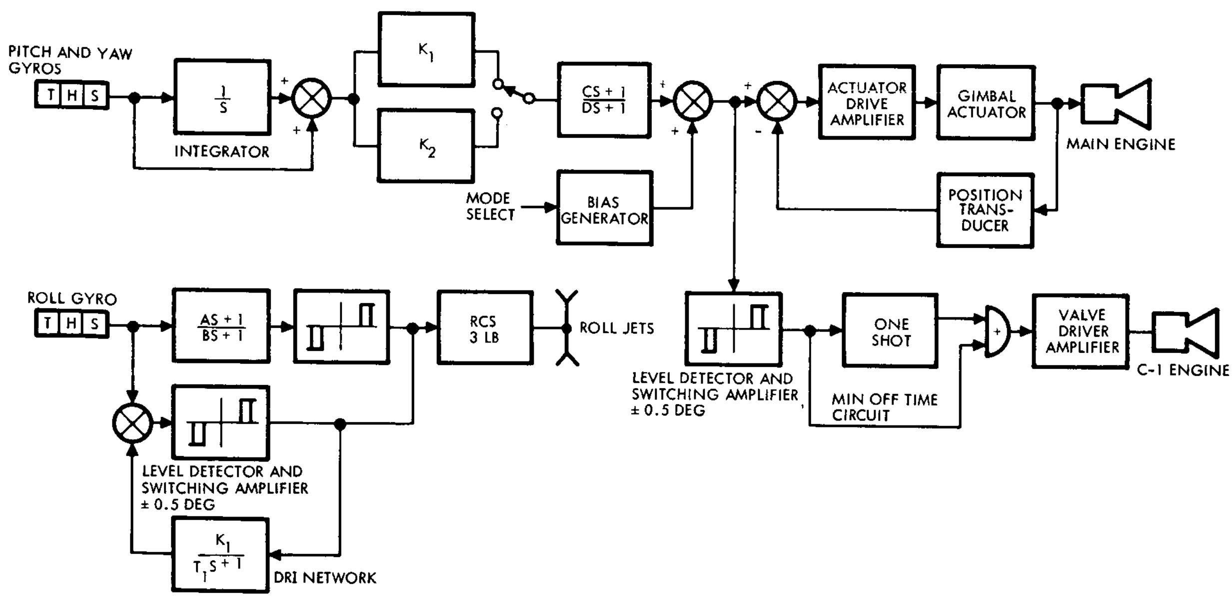


Figure 5-7
THE THRUST-VECTOR CONTROL MODE of operation of the guidance and control subsystem involves this arrangement of components

engine operation. Pulsing "off" of these engines is accomplished to maintain attitude control. The actuator commands are routed to a switching circuit with minimum "Off-time" control to ensure proper operation with these engines. Attitude rate commands are employed for this backup powered flight operation in order to achieve the desired Mars orbit.

The high thrust cold gas jets are used for stabilization about the roll axis, using outputs from the roll gyro. Rate damping in pitch and yaw is provided by passive lead networks operating on the gyro signals.

A gain change from high to low thrust operation is indicated on the block diagram. The high thrust (9850 pounds) is used during the orbit injection mode; the low thrust (1700 pounds) is used for midcourse and orbit trim velocity corrections. The thrust variation imposes the necessity to change gains for vehicle stability and control.

The total thrust vector pointing error during these powered flight phases will be less than 0.43 degree. Use of electrical and mechanical engine biases, attitude feedback integration, spacecraft attitude alignment, and celestial sensor signal calibration of the gyros all aid in reduction of the errors.

The velocity change during powered flight is sensed by a velocity meter which consists of a linear accelerometer, DC amplifier and AC converter. The velocity meter has a proportionality accuracy of 0.1 percent. The quantization level of the velocity increment is 0.015 ft/sec/pulse. The velocity increment pulses are counted by the computer and sequencer during the engine operation; when the desired velocity increment is reached, the engine will be shut down automatically by command. Upon completion of the powered flight, the celestial references are reacquired either by commanded turns, as described for the inertial hold mode, or by switching to the sun and Canopus acquisition modes of operation.

5.2.2.11 Photo-Imaging Mode

This mode of operation is essentially identical to the cruise mode except that the deadband is reduced from ±0.5 to ±0.25 degree to allow more accurate directional control of the planetary scan platform.

5.3 COMPONENT DESCRIPTION

5.3.1 Inertial Reference Unit

The inertial reference unit consists of three single-degree of freedom gyros and a force-balance accelerometer, each with an associated electronic caging loop. The package provides angular rate or angular position information in three axes and indicates velocity increments along the roll axis; preliminary specifications are given in Figure 5-8.

Reliability was the overriding consideration in inertial reference unit design.

After a thorough survey of the available instruments, the Nortronics GI-T2-D was selected as potentially the most reliable gyro for this application. Except for minor design changes, this gyro is identical to the GI-T1-B utilized in the Minuteman II and III inertial guidance systems. Over 850 of these instruments have been manufactured and extensive field test data is available to substantiate their reliability. In addition, this unit has demonstrated stable gas spin-bearing operation at zero-g.

The accelerometer choice was made on a similar basis. The Kearfott 2401 accelerometer was selected because over 5000 have been fabricated and tested. Four-year stability data confirms its capability for the Voyager application.

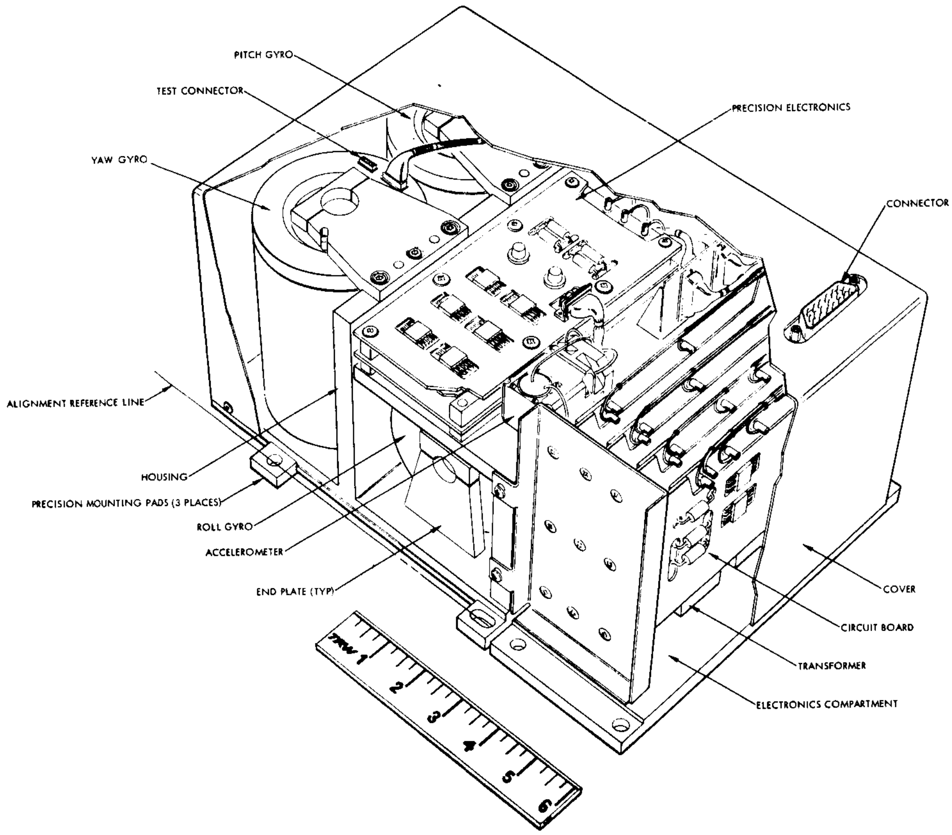
The functional block diagram of the Inertial Reference Unit appears in Figure 5-9. The position gyro loop is of the conventional autopilot type. Torque-to-balance loops are used to initially cage the gyro gimbals to their pickoff nulls, and also to provide angular rate information.

Angular position information is obtained from the gyros by opening the caging loops and utilizing the demodulated pickoff signal. When in the angular position mode, control system commands as required to maneuver the spacecraft are generated by passing a precision current through the gyro torquer. Inertial stabilization of the spacecraft is obtained by removing the electrical torquing currents from all gyros.

When the inertial hold mode is initiated the attitude error due to the limit-cycle error from the initial position of the instantaneous position

PRELIMINARY SPECIFICATION

Inertial Reference Unit



Purpose

Provides angular rate or position signals in the spacecraft pitch, roll and yaw axes using three body-fixed floated rate integrating gyros. Gyros can be torqued to provide maneuver capability or operated in position hold mode. During acquisition gyros are operated in rate mode to provide damping signals. Provides a measurement of incremental acceleration along the thrust axis using a single force-balance linear accelerometer.

Performance Characteristics

Gyro input angle:	±10 deg
Gyro torquing rate:	± 0.1 or ±0.2 deg-sec ±1000 rpm 3σ
Gyro drift:	
Acceleration insensitive	0.1 deg/hr, 3σ
Acceleration sensitive	0.3 deg/hr/g, 3σ
ACCELEROMETER scale factor:	0.015 ft/sec/pulse ±121 ppm 3σ
ACCELEROMETER bias:	$< 1 \times 10^{-4}$ g

Physical Characteristics

Size:	7 x 8 x 12 in.
Weight:	25 lb
Power DC	40 watts minimum 54 watts maximum (including heaters)

Figure 5-8

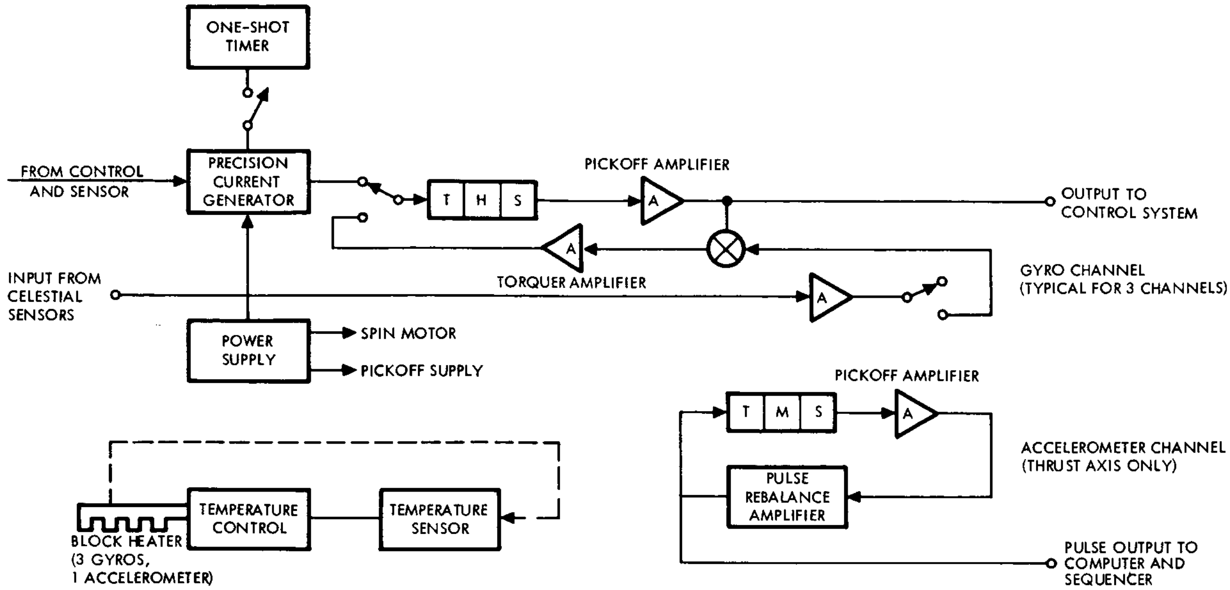


Figure 5-9
INERTIAL REFERENCE ASSEMBLY includes gyros, accelerometer and associated torquing, power supply and temperature control circuitry.

of the spacecraft in the limit cycle is reduced by using the optical sensor signals to provide a reference correction in the following manner. Prior to entering the attitude hold mode or any of the maneuver modes, signals from the Canopus and the pitch and yaw sun sensor channels are summed with the rate loop signals of the roll, pitch and yaw channels, respectively. This establishes a gyro gimbal angle and its associated pickoff signal equivalent to the instantaneous error angle between the particular gyro input axis and the corresponding optical sensor line of sight.

When the system is switched to the gyro position mode (spacecraft inertial hold or maneuver mode), the gyro caging loop is opened and the optical sensor signals are removed. This causes the optical sensor error angles to be stored in the corresponding gyros, establishing the gyro reference axes parallel to the optical lines of sight and independent of the deadband within which the spacecraft attitude is controlled.

The precision current source in the inertial reference unit is switched on command into the roll axis, the pitch axis and the yaw axis. The torquing current from the precision current supply is commanded by the spacecraft command sequencer for repositioning the spacecraft about any axis at a precise rate of ±0.2 deg/sec. Provision is also made to roll the spacecraft in ±2 degree increments. The inertial reference unit contains a roll one shot circuit to perform this maneuver. Upon command this circuit applies the precision current to the roll torquer for 10 seconds. A slew polarity command must also be provided. These inputs cause the current from the precision current source to torque the gyro at ±0.2 deg/sec. After 10 seconds the spacecraft has moved 2 degrees, at which time the roll one-shot countdown stops the motion.

Provision is made for in-flight self-test of the gyro loops without special circuitry. An appropriate command signal places the gyro loops in the rate mode. The roll, pitch, and yaw output signals are read out through the normal telemetry channels, and should be essentially zero at this time. The loops are then opened by commanding the inertial hold mode and pitch, roll and yaw slew rates are inserted into each channel in sequence for a 20-second time period. This produces an output signal proportional to an input angle offset of 4 degrees in each channel. These output signals are read out through the normal telemetry channels. The change between the output with no input and that with the simulated input demonstrates that the gyro and its loop and the torquing electronics are operating properly.

The two inertial reference units, normally off during the cruise modes, are energized for the powered flight phases, reorientation maneuvers, and inertial hold operations during periods of celestial occultation. During the reorientation maneuver, gyros in both units are torqued and the error signals compared to provide additional verification that a satisfactory maneuver has been accomplished. During the powered flight phases, an automatic switching to the redundant set of gyros can be accomplished based upon logical operations with the gyro error signals. If the gyro error and differences between gyro errors are greater than predetermined levels, the gyro switch would be made. The error levels would be established from simulation studies employing stability and structural constraints. Also, the outputs of the gyro spin motor rotation detectors would be employed in the gyro selection logic.

The Kearfott 2401-series force balance accelerometer operates in a pulse torque rebalance loop. The output of the accelerometer signal generator controls the pulse rebalance electronics, which produce a train of positive and negative torque pulses of equal amplitude and length to servo the pickoff output to null. The information output of the pulse balance electronics is a pulse train where each pulse represents an increment of velocity gained along the spacecraft axis. The time duration between pulses is a function of the spacecraft acceleration. A minimum and maximum duration can therefore be set in a failure detection logic circuit to enable switching to the redundant accelerometers.

The accelerometer loop self-test is accomplished by applying a current obtained from the accelerometer self-test circuit to the accelerometer torquer for a preset time. Initially, the circuit is monitored with no input disturbance by counting the output pulses in the computer and sequencer for a fixed time interval. This establishes a reference and checks for gross bias errors. Then the simulated input is switched in, causing the loop to produce a canceling current which counteracts the disturbing current. A pulse train associated with the loop generated cancelling current appears at the normal accelerometer output. These pulses are then counted in a register located in the computer and sequencer and are subsequently read out through the normal operational telemetry channel. These output and measurements demonstrate that the accelerometer and its loop are operating properly.

Two functions are monitored continuously through the telemetry channels. These are the temperature sensor and the spin motor rotation detectors on all three gyros. These are normal housekeeping functions to enable detection of faults. The temperature sensor is the same one used for the temperature control loop. The spin motor rotation detectors operate from a pickoff in each gyro which provides a signal whose frequency is proportional to the gyro wheel speed.

The inertial reference assembly consists basically of two separable subassemblies: a temperature-controlled housing assembly for the inertial sensors and the precision electronics, and an electronics compartment.

The housing is precisely machined aluminum alloy framework with a reference plane defined by three precision mounting pads. In addition, precision holes through two of these pads define a reference line and this combination establishes the package coordinate system. The gyro mounts consist of three pairs of end plates attached to the housing and line-bored with respect to the reference line and plane to establish one component of input axis alignment. The other component of gyro input axis alignment is established by indexing each gyro alignment pin against the precision stop on one of the end plates. The accelerometer mounts on a carefully machined surface which is parallel to the housing pads and hence establishes the accelerometer input axis parallel to the spacecraft roll axis.

5.3.1.1 Gyros

The GI-T2-D gyro recommended for the Voyager spacecraft contains minor modifications of the proven GI-T1-B design. Specifically, the fluid viscosity is increased to lower the gain from 6 to 0.3. The output axis angular freedom is thereby increased from ±1 arc minute to ±3 degrees, and the torquer scale factor is increased from 0.2 deg/hr/ma to 216 deg/hr/ma. The basic gyro structure, including the gas bearing, float and case parts remain unchanged. Its characteristics are shown in Table 5-4.

Table 5-4. GI-T2-D Gyro Characteristics

Spin Motor:	800 cps, 2ϕ, 6.5 watts
Torquer Scale Factor:	0.006 deg/sec/ma
Angular Momentum:	1.8×10^6 gm/cm^2/sec
Input Freedom:	±10 degrees
Gain:	0.3
Fixed Drift Stability:	0.1 deg/hr, 3σ
G-Sensitive Drift Stability:	0.3 deg/hr/g, 3σ
Size:	3.5 x 5 in.
Weight:	3.5 lb
MTBF:	500,000 hr

5.3.1.2 Accelerometer

The characteristics of the 2401 force balance pendulous accelerometer are shown in Table 5-5. As well as having proven reliability, its other performance parameters and its size, weight and power show it to be equal to, or better than, competitive instruments.

Table 5-5. Accelerometer Instrument Performance Characteristics

Bias:		1×10^{-4} g
Bias Stability:	Random:	90×10^{-6} g
	Systematic:	50×10^{-6} g, 3σ for 1/2 year
Threshold:		$<1 \times 10^{-6}$ g,
Resolution:		$<1 \times 10^{-6}$ g
Size:		2 x 1 x 1.3 in.
Weight:		0.4 lb
MTBF:		1,000,000 hrs

5.3.1.3 Gyro Loop/Electronics

The gyros are operated open loop in an autopilot configuration while in the position mode. In the rate mode the gyros are operated with conventional electronics in a closed loop with a steady state rate scale factor of 16.6 volts/deg/sec and with a loop time constant of 7.6 seconds. A rate threshold of 0.01 deg/sec translates into a voltage of 0.166 VDC at the output terminals. The electronic amplifier limits its current to ±67 milliamperes corresponding to a maximum input rate, requirement of 0.4 deg/sec. However, the operating gyro/circuit combination will withstand input angular rates without damage up to a maximum rate of 4 rad/sec. about any gyro axis.

5.3.1.4 Accelerometer Loop Electronics

The accelerometer is pulse rebalanced to provide a direct digital output consisting of pulses which carry a weight of 0.015 ft/sec per pulse. The loop is scaled for a maximum input of 2.8 g and is operated at a pulse repetition rate of 6400 pps.

Under acceleration the accelerometer pendulum turns about its axis. The accelerometer pickoff senses this displacement from the null position and conveys its magnitude to the electronics as a double-sideband suppressed-carrier signal. This is amplified, synchronously demodulated, converted to current pulses, and applied to the accelerometer torque motor to return the pendulum to its null position. The pulse amplitude is controlled by means of a precision current regulator and the width is determined by the 6400 pps clock spacing so that each pulse represents a fixed acceleration-time integral or velocity increment.

5.3.1.5 Spin Motor Rotation Detector

A spin motor rotation detector of the magnetic-slug type gives a positive indication for each channel that the gyro wheel is running at synchronous speed. Due to synchronous motor hunting, instantaneous speed measurement is undesirable so that a time averaging scheme utilizing a spin motor pulse counter and a reference counter together with the appropriate logic is used.

The inertial reference assembly is temperature controlled using a proportional electronic controller and a block heater. The controller has a set point variation of 0.5 ^{o}F to reduce gyro and accelerometer gain changes and dissipates a maximum of 20 watts.

5.3.2 Coarse Sun Sensor

The coarse sun sensor provides analog signals that locate the sun over a 4π-steradian field of view. The preliminary specifications appear in Figure 5-10. The system is essentially the same as that used on the Orbiting Geophysical Observatory. The coarse sun sensor consists of four similar assemblies, bracket mounted at four locations on the periphery of the solar array as shown in Figure 5-11. The locations of the sensors are chosen so as to provide a full 4π-steradian field of view without interference or shadowing from antennas and other appendages. In order to accomplish this, the fields of view of the individual sensors are restricted by shades mounted to the brackets.

Each sensor contains two flat silicon photovoltaic cells, one of which provides yaw sensor signals, while the other provides pitch signals.

PRELIMINARY SPECIFICATION

Coarse Sun Sensors

Purpose

The coarse sun sensors provide output signals indicative of the location of the sun in a 4π steradian field of view. They are arranged and interconnected so as to generate control signals of the proper magnitude and polarity to drive the spacecraft in a direction such that the sun will appear in the field of view of the fine sun sensor.

Performance Characteristics

Field of view (for all coarse sensors) 4π steradian

Equivalent pointing accuracy $\pm$ 1 degree

OUTPUT SIGNAL

Pitch $= K \sin\theta \sin\phi$

Yaw $= K \sin\theta \sin\phi$

where ϕ = sun aspect angle

and θ = sun rotation angle

Physical Characteristics

Size 1-1/2 x 3/4 x 3/4 in.

Cells and thermal control block

Brackets 2-1/2 in. long

Power 300 mw average

Weight 0.6 lb per unit for 4 units

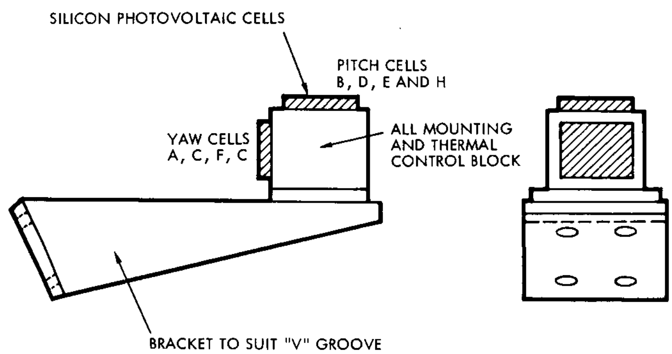


Figure 5-10

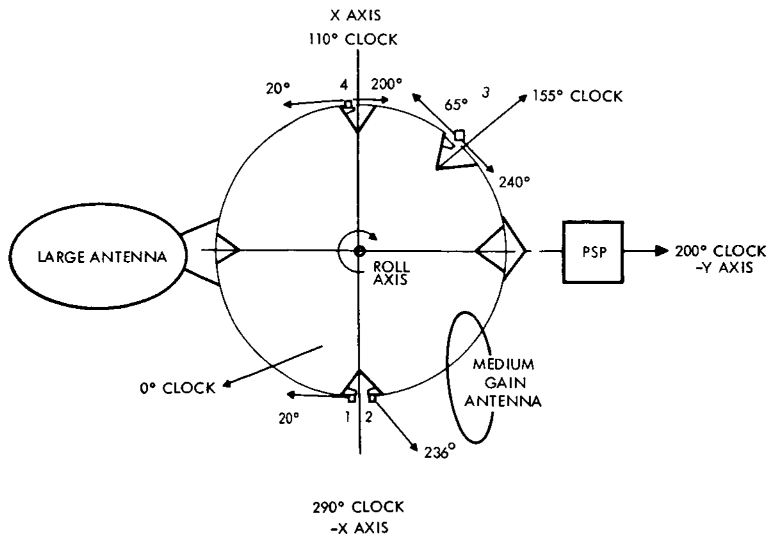


Figure 5-11

LOCATION OF COARSE SUN SENSOR shows that the fields of view of each add together to give full spherical coverage. All angles shown are clock angles.

The location of the individual cells is shown in Figure 5-12, while the individual sensors are shown in Figure 5-10. The coarse yaw signal is obtained by electrically adding the outputs of the yaw cells, and the coarse pitch signal is derived by electrically subtracting the signals of the pitch cells, as shown in Figure 5-13. The pitch amplifier is designed to keep the pitch signal output within the same range as the yaw signal.

The units are positioned so that if one assembly is in shadow, the other is illuminated. A preliminary study indicates reflections can be kept from entering the detectors by placing sun shades near each assembly. Typical shade dimensions are 1-1/2" x 3".

5.3.3 Fine Sun Sensor

The fine sun sensor consists of a silicon photovoltaic quad cell mounted on a quartz block behind a shadowing mask as shown in Figure 5-14. Its Preliminary Specification is given in Figure 5-15. The pitch

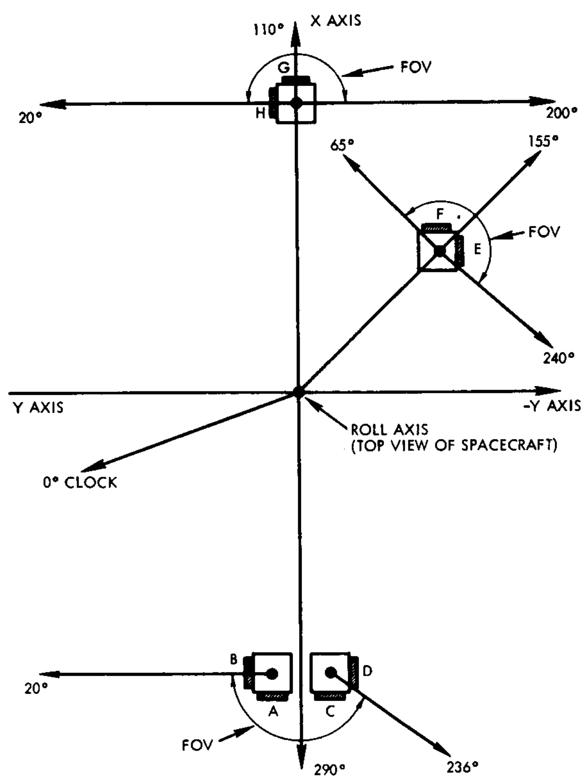


Figure 5-12
LOCATION OF COARSE SUN SENSORS showing fields of view and position of cells on each sensor. Angles shown are clock angles, cells are shown edge-on.

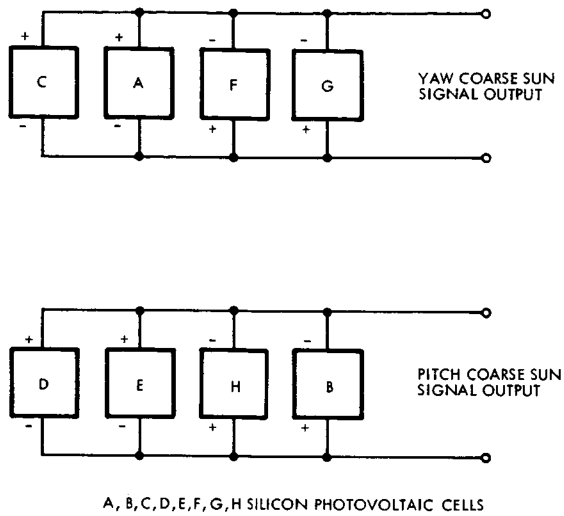


Figure 5-13
METHOD OF INTERCONNECTION of coarse sun sensor photovoltaic cells to obtain signals of proper polarity for pitch and yaw control.

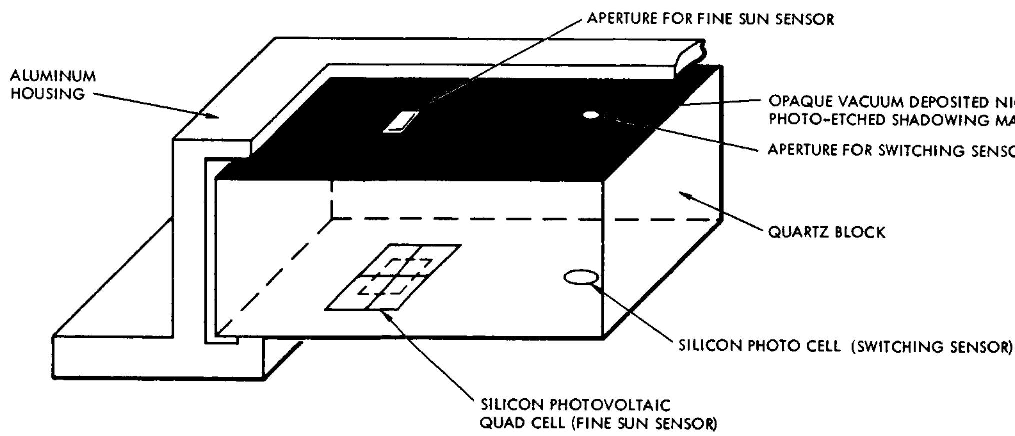


Figure 5-14
THE FINE SUN SENSOR-SWITCHING SENSOR ASSEMBLY makes use of vacuum deposited shadowing mask.

PRELIMINARY SPECIFICATION

Fine Sun Sensor and Switching Sensor

Purpose

The fine sun sensor provides two axis output signals of the proper polarity and magnitude to enable pointing of the negative roll axis of the spacecraft to the sun. Combined with the fine sun sensor is a switching sensor which when illuminated generates a signal which transfers attitude control of the spacecraft from the coarse sensor to the fine sensor. The switching sensor is illuminated only when the sun is in the field of view of the fine sun sensor.

Performance Characteristics

Field of view	± 10 deg
Linearity	± 10%
Null accuracy and stability	± 0.1 deg 3σ

Output signal

Pitch = $K \sin\theta \sin\phi$

Yaw = $K \sin\theta \cos\phi$

where θ = sun aspect angle

ϕ = rotation angle

Physical Characteristics

Size	2 x 3 x 2 in.
Weight	0.2 lb
Power	700 mw average

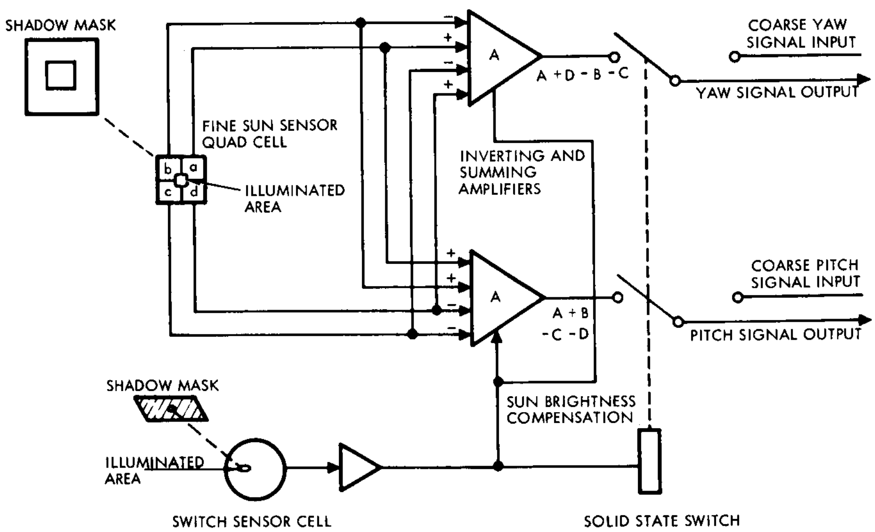


Figure 5-15

(P) and yaw (Y) signals are obtained by electrically adding the signals from adjacent quad cell quadrants in pairs and then electrically subtracting the pair signals. The nominal output signals have the form (see Figure 5-16

$$P = K \sin\theta \sin\phi$$

$$Y = K \sin\theta \cos\phi$$

where K is essentially constant. $\sin\theta$ is constant over the range $\theta = 0$ to 10 degrees.

Although linearity will be maintained throughout the mission, scale factors will vary. The major cause of scale factor variation is the decrease in solar intensity by a factor 2.6 over the range of the mission. It is possible to use the switching sensor signal in an automatic gain control circuit to compensate the fine sun sensor for this change of solar intensity.

A signal to switch from coarse to fine sun sensor can be obtained by adding the signals of all four quad cell quadrants, but this may cause considerable signal discontinuity. To minimize this, a separate sensor

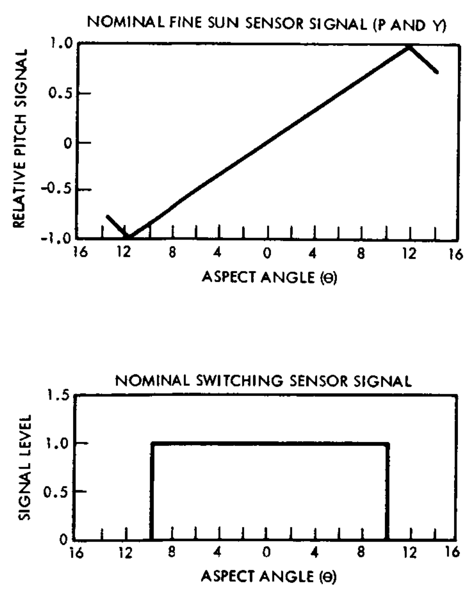


Figure 5-16
OUTPUT SIGNAL CHARACTERISTICS OF THE FINE SUN SENSOR is a ramp function while that of the switching sensor is a step function.

is used. It is mounted on the same quartz block as the fine sun sensor such that coarse-fine switching occurs when the coarse sun sensor signal is within 20 percent of the fine sun sensor signal.

The fine sun sensor-switching sensor assembly consists of a single quartz block 1-1/2 x 3 x 1 inch mounted inside of an aluminum housing bolted to the spacecraft. A shadowing mask is located on one face of the quartz block, and the detectors on the opposite face. The shadowing mask is formed by photo-etching a vacuum deposited nickel film.

5.3.4 Canopus Sensor

The Canopus sensor design chosen for the Voyager Program utilizes an electronically-scanned image dissector and is similar to that used in the Lunar Orbiter and Mariner spacecraft. However, several important design improvements have been incorporated to make the sensor more accurate, more reliable, and more resistant to stray light impingement. The last item is extremely important since both the Lunar Orbiter and Mariner proved quite susceptible to glint from exposed spacecraft parts. A photo of a prototype TRW Canopus sensor is shown in Figure 5-17.

The preliminary specification for the Canopus Sensor appears in Figure 5-17.

The image dissector photocathode converts the optical star image to an electron image which is then accelerated down the drift tube towards the aperture plate. Deflection coils symmetrically positioned around the drift tube cause the electron image to be swept back and forth across the aperture, forming a video signal which is both pulse position modulated and pulse length modulated with respect to the master timing pulses. The 1800 volt image dissector supply comes from a DC to DC converter, free running at approximately 3200 Hz. This frequency acts as the master timing pulse for the entire tracker; therefore, any transient spikes caused by high voltage switching occur at times that do not interfere with the tracking signals.

In the acquisition mode, the acquisition scan circuits provide a raster scan which causes the 1 degree by 1 degree instantaneous field of view to be digitally stepped to cover a 4 degree by 30 degree field. The

PRELIMINARY SPECIFICATION

Canopus Sensor

Purpose

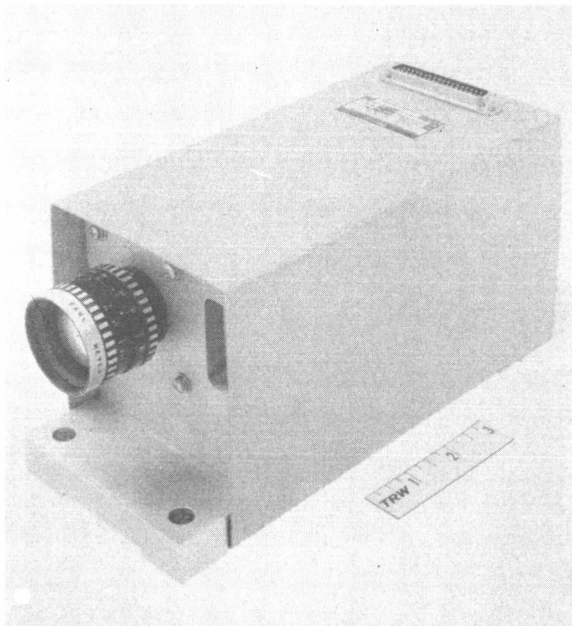
Sense the position of the Star Canopus and provide error signals for roll control. Provides electronic gimballing capability in two axis and adjustable threshold.

Performance Characteristics

Acquisition field of view:	$4^\circ \times 30^\circ$
Instantaneous field of view:	$1^\circ \times 1^\circ$
Gimballing:	All electronic in both axis Closed loop feedback
Sensitivity:	Brighter than $+0.5\ m_v$
Threshold:	Adjustable on command to nominal 0.6, 0.4, and 0.3 canopus
Noise equivalent angle:	15 arcseconds rms
Time constant:	0.3 second ±0.5 second
Optics:	20 mm F/1.0 refractive

Physical Characteristics

Size:	4 x 12 in.
Weight:	7 lb
Power:	6 watts (peak)



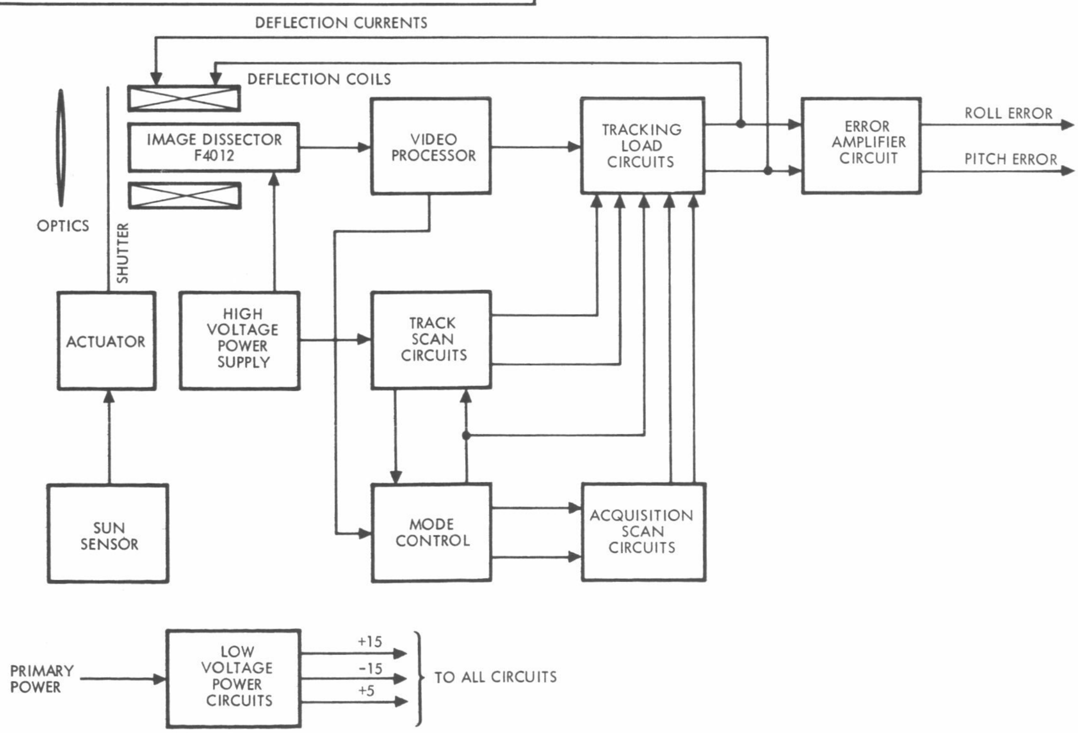


Figure 5-17

minimum steps per frame would be 4 steps by 30 steps with no overlap. To give overlap and simplicity the raster was made 8 steps by 32 steps. This gave sufficient overlap and assured a solid signal for positive acquisition. With the step rate set by the master timing pulses of 3200 Hz, the frame rate is 0.08 second.

The 3200 Hz from the high voltage supply is gated to the digital-analog converter (roll) through a diode AND gate that is closed when a star pulse is received from the mode logic circuitry. The roll digital-analog converter is a three-stage integrated-circuit counter. The roll stair-step sweep is formed by switching precision resistors with the outputs of the integrated circuit flip-flops. The output of the last flip-flop in the roll digital-analog converter forms the input to the pitch digital-analog converter, which is a five-stage integrated circuit counter. The pitch stair-step is formed in the same manner as the roll stair-step. The stair-steps are then used to drive the respective tracking loop summing amplifiers.

Once video pulses have been received, the mode logic forces the tracker to stay in the acquisition mode for six complete rasters which allows the video processing circuit to select only the brightest star above the commanded threshold. (The threshold can be adjusted upon command to 0.6, 0.4, and 0.3 Canopus intensity.) When in the track mode, the mode control circuits cause the sweep waveform to switch from the raster to the cross-scan pattern and close the tracking loop.

The track scan circuit generates a triangular waveform which is sequentially switched between the two deflection coils to provide a cross scan sweep pattern for tracking the star image on the photocathode. They also provide the synchronizing waveform for the demodulation in the tracking loop circuits.

The signal from the image dissector is amplified in the video processor and is provided as a shaped pulse to the tracking loop. The video processor also provides a star magnitude signal, and sets the threshold levels. A star presence signal is generated for the mode control circuits.

The tracking loops, one in each channel, consist of a synchronous demodulator, tracking integrator, summing amplifier and deflection coil driver. A field effect transistor shorting bridge on the integrator prevents the integrator from drifting in the acquisition mode. To assume zero drift operation, the demodulators are gated off at the same time as the shorting bridges.

The inputs to the scanning amplifiers are the demodulator output, the track integrator output, the tracking scan waveforms and the acquisition scan waveform. The summing amplifier drives the deflection coil driver which is coupled to the deflection coils and the error amplifier circuits.

The signals to deflection coils are synchronously detected to remove the track sweep, filtered in a low pass filter and amplified to provide the desired error signal gradient.

A shutter protects the image dissector prior to stabilization to prevent the sensor looking directly at the sun or at the sunlit earth. A sun sensor consisting of a cadmium sulphide cell located behind a field stop with a field of view of ±40 degrees actuates a shutter when the sun is within this angle. The shutter consists of a rotary solenoid with a thin metal flag. The sensitivity of the circuit is adjusted so that when the sunlit earth, or any object bright enough to damage the photocathode, lies within the 40 degrees field of view of optical system, the shutter will automatically close.

To prevent glint interference, a large protective shade in front of the basic sensor has been designed to allow for maximum tracking field of view with a stay-clear area of only 20 degrees by 40 degrees.

The lens used in the Canopus sensor is the 20 mm F/1.0 refractive optics, Part No. 34400, manufactured by the Pacific Optical Company and space qualified on the Lunar Orbiter Canopus Star Tracker.

The detector for the Canopus sensor is a miniature image dissector, the F4012 with an S-20 photocathode. Figure 5-18 is a cutaway view of the image section of the tube. Since the F4012 is a "true" image dissector with a drift tube in the image section, it has better focusing and image deflection capabilities than does a pseudo dissector such as the FW-142, FW-143, or the CL 1147.

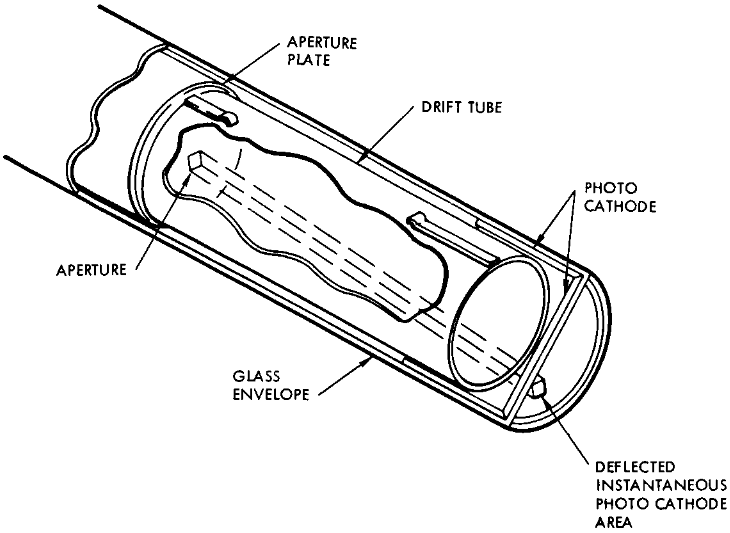


Figure 5-18
THE F4012 IMAGE DISSECTOR is a true image dissector with a drift tube in the image section. This provides it with superior image deflection and focusing capabilities.

The F4012 is a magnetically deflected and magnetically focused tube, which allows fine alignment adjustments, impossible with the CL1147 and its offspring. The coupling of the F4012 anode signal to the video amplifier is accomplished with low-voltage space-qualified capacitors rather than with high voltage disc caps as is the case with the CL1147. Also, the F4012 has less than a 10 percent deviation of photocathode current over the face of the tube, while the CL1147 shows greater than 50 percent variation.

Maximum use has been made of integrated circuits in the design of TRW's Canopus sensor. For instance, the choice of a digital raster for acquisition has made it possible to use integrated circuits exclusively for this function. The flip-flop used are Fairchild LPDTμL 9040 low power units, feeding Sprague UD4035 buffer switches, which in turn feed Sprague UD4036 ladder switches and Sprague UT1001 ladder adders. This allows a very high density packaging with very high reliability. The linear circuits utilize Fairchild μA709's or National's LM101.

The rotary solenoid used for the sun shutter has been spec qualified and is manufactured by DACO Instrument Company as Model D-EL-6.

A cutaway view of the method used to support the tube, optics and sun shutter is shown in Figure 5-19. This technique has been successfully used for other space qualified sensors and the use of the triple layer co-netic shield in the supporting element for the optics and tube eliminates many vibration alignment problems.

Alignment of the Canopus sensor with the inertial reference unit is by means of an optical mirror mounted to the face of the sensor. The boresight axis of the sensor is aligned in the laboratory to be normal to the mirror, and thus the mirror can be auto-collimated with the inertial reference unit prior to spacecraft installation.

5.3.5 Limb and Terminator Detector

A Preliminary Specification for the Limb and Terminator Detector is shown in Figure 5-20. Whenever reflected sunlight from the partially illuminated planet enters the 2 degree (cone) x 100 degree (clock) field of view of the sensor, the resistance of either or both of the cells CDS1 or CDS2 changes in accordance with the level of incident light. A change in resistance, from the value when the cell is dark, offsets the balance of the bridge network and results in a signal input to the amplifier. The signal is amplified, filtered, and is then processed through a threshold detector. A positive output of the level detector indicates that the sunlit portion of the planet is within the field of view of the sensor. Zero output

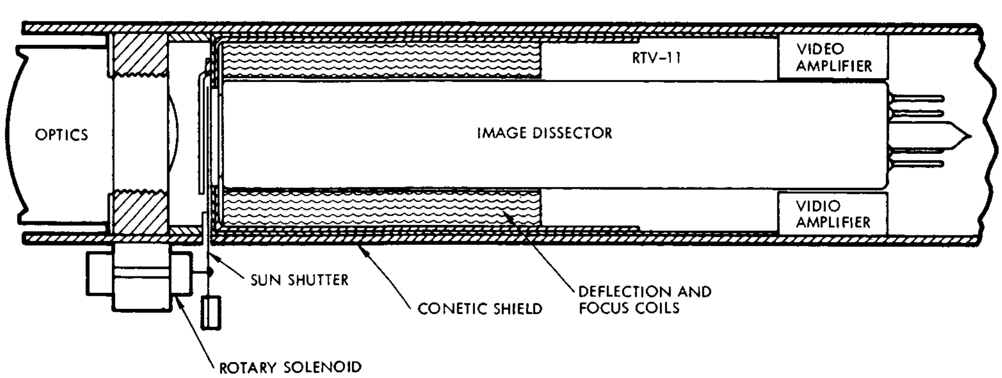


Figure 5-19
SUPPORT OF THE IMAGE DISSECTOR TUBE AND OPTICAL PACKAGE in this way has been successfully used on other space qualified sensors. Use of the triple layer conetic shield for support eliminates many vibration alignment problems.

PRELIMINARY SPECIFICATION

Limb and Terminator Crossing Detector

Purpose

Provides output signals indicative of the crossing of the Mars limb and terminator for the purpose of sequencing the scientific experiments. Detects the appearance of the sun illuminated limb and the dawn or evening terminator in 90 degree cone angle plane in spacecraft coordinates.

Performance Characteristics

Field of view (2 detectors mounted 180 clock deg apart are used)	± 50 deg clock angle x ± 1 deg cone angle
Orbital altitude range	500 KM to 5000 KM
Accuracy	± 1 degree
Optics	35 MM dia f/2
Spectral band	0.45 to 0.7 microns

Physical Characteristics

Size	4 x 1.5 x 2.5 in.
Weight	0.6 lb
Power	200 mw

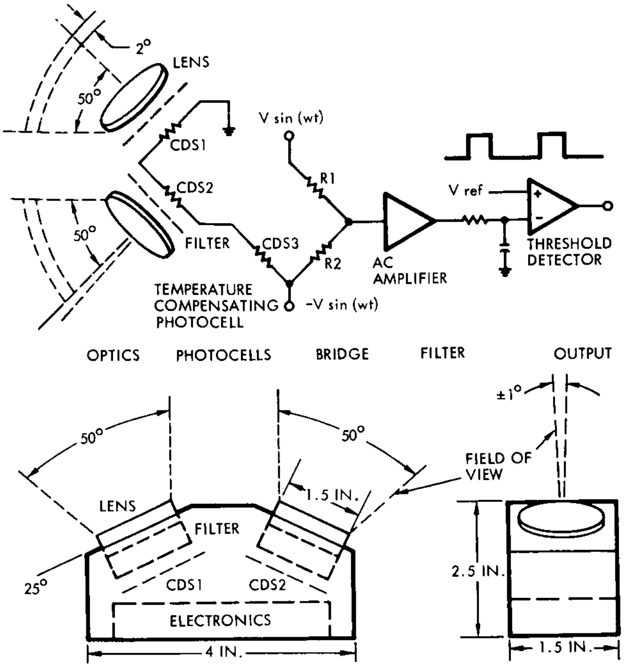


Figure 5-20

level indicates space or the dark portion of the planet is in view. Transition of the output between these two levels occurs as the sensor field of view crosses the planet limb and terminator planes.

The sun-planet-sensor geometry is shown pictorially in Figure 5-21. The luminous flux incident on the photo-cells may be calculated by summing the contributions of light reflected from the many elements of area dA within the field of view of the sensor. The reflection properties are assumed to follow Lambert's Law.

Figure 5-22 shows the flux magnitude as a function of the various parameters. The sensor error may be obtained by observing the range of spacecraft-planet angles, at which the threshold level is crossed. Choosing a threshold level of 3.0×10^{-4} lumens, results in an error range of ±0.6 degree for the worst case parameter extremes shown in this figure.

The threshold level of 3.0×10^{-4} lumens would be reached if the total cell area (9×10^{-4} ft^2) were illuminated at a luminous intensity of 0.33 footcandles. This magnitude is well within the operating range of a cadmium sulphide cell.

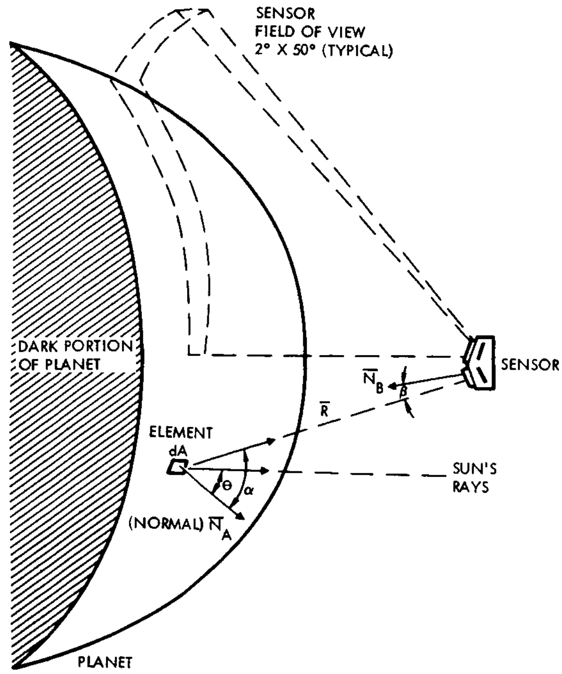


Figure 5-21
SUN-PLANET-SENSOR GEOMETRY enables sensor to provide an unambiguous limb or terminator signal.

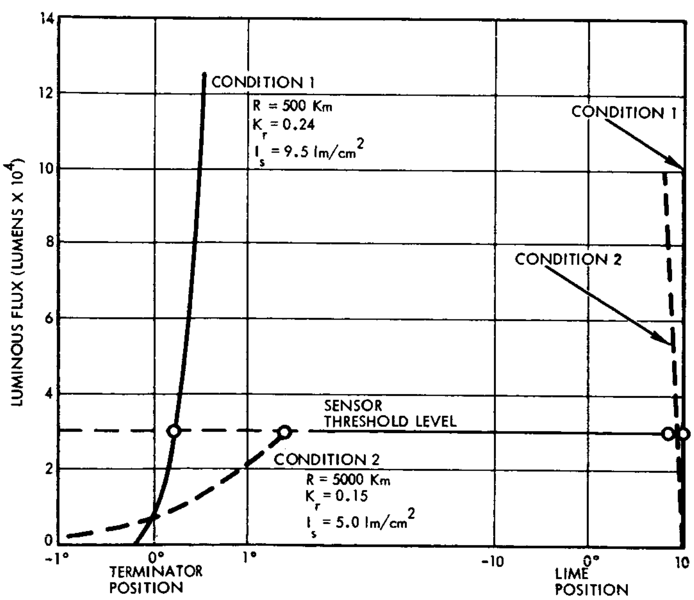


Figure 5-22
LUMINOUS FLUX INCIDENT UPON PHOTOCELL of the limb and terminator crossing detectors for several orbital conditions.

Two assemblies are mounted on the spacecraft with the fields of view to be oriented for coverage at 90 degrees ±50 degrees and 270 degrees ±50 degrees in clock angle, and 90 degrees ±1 degree in cone angle.

5.3.6 Reaction Control Subsystem

The reaction control subsystem provides the control torques required for spacecraft attitude control. Its preliminary specifications appear in Figure 5-23.

The impulse requirements for the recommended Voyager configuration using the 5000 lb capsule are:

Cruise:	307 lb-sec
Acquisition and Maneuver:	739
Powered Flight Roll Control:	130
Total Impulse Required:	1176 lb-sec

The recommended reaction control system is a completely redundant stored nitrogen gas system, operating at two thrust levels and incorporating electrical resistance heating of the gas immediately upstream of the 12 nozzles. The two redundant tankage and feed systems each supply two identical sets of six nozzles, i.e., each set consists of one each plus and minus pitch, plus and minus yaw, and plus and minus roll nozzles. In this way half of each couple is supplied from one system and half from the other. The valves, nozzles, and heaters are capable of operating at both the high and low thrust levels.

The design of the reaction control system insures that a failure of any particular component does not jeopardize the overall spacecraft mission, although reaction control system performance may degrade. Among the more important failure modes considered are a control valve stuck open, causing a disturbing torque as well as depleting the propellant, and a leak in the propellant storage tank causing a depletion in propellant. In this last case, leakage caused by meteoritic impact rupture is taken into account, as well as leakage through mechanical joints.

Each thruster assembly consists of a heat exchanger, resistance heater and the two (or four) nozzles. As shown, two (or four) gas feed lines enter the thruster along with a two-wire heater lead connected to the heater element.

PRELIMINARY SPECIFICATION
Reaction Control System

Purpose

Provides control torques required to achieve and maintain spacecraft attitude during acquisitions, cruise and maneuvers. Also provides torques for roll control during operation of the main or C-1 engines.

Performance Characteristics

Working fluid	Cold gaseous nitrogen, electrically heated during cruise
ISP	60 (120 when heated)
Thrust level	3 or 0.2 lb, selectable
Minimum impulse bit	5.0×10^{-3} lb-sec
Torque lever arm	10 ft
Total required impulse	1176
Storage pressure	3000
Working pressure	630 psia for 3 lb thrust, 39 psia for 0.2 lb thrust

Physical Characteristics

Gas weight	61.1 lb (includes 3X for redundancy ullage, leakage, etc.)
Tank weight	56.0 lb total for two tanks
Other component weight	41.2 lb
Total system weight	158.3 lb
Tank diameter	19.1 in.
Power requirements	
Solenoid valve	25 watts each
Heated thruster	
Quad heated thruster	16 watts each
Dual heated thruster	12 watts each

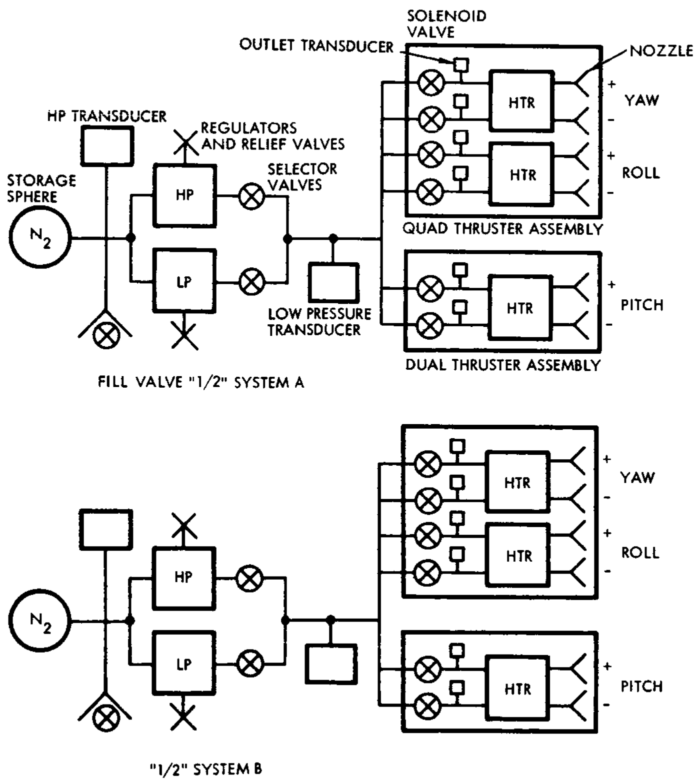


Figure 5-23

The thrust level is controlled by operation of the regulator selection valves. With one valve open the 630 psia regulator is activated, supplying gas at this pressure to the solenoid valves and nozzles. With this valve closed and the other open, the 39 psia regulator supplies the system at this pressure. The nozzles are designed so as to provide 3 pounds thrust at the high pressure and 0.2 pound thrust at the low pressure. If a selector valve fails to open, the parallel system can maintain operation in a slightly degraded manner. Should a nozzle feed valve fail to close, the disturbance torque from this failed valve will cause the opposite direction valve couple to actuate. The average flow rate through each valve of the couple will be one-half the flow rate through the failed valve in order to maintain spacecraft control. Since one valve of the couple is in the half system with the failed valve, the flow rate from the failed half system is three times the flow rate of the redundant half system. To protect against this type of failure at the beginning of the mission and still carry enough gas to complete the mission in the remaining system, 1.5 times the mission required impulse must be carried in each tank. One-third of this gas is expelled in the event of a failure leaving 1.0 times the mission requirement in the tank when the other half system is completely expelled. Following such a failure, operation in all axes will be limited to one-nozzle torques rather than pure couples.

The line length between the nozzle feed valves and the thruster assemblies is kept to a minimum, since short pulses in long lines lead to metering of the gas flow at the control valves rather than at the thruster assemblies. This would not allow the flow control necessary to obtain the higher impulse available in the heated gas system. If detailed analysis shows that the temperatures of the valves can become too low, foil-type low powered heaters (3 watts each) will be provided to maintain valve temperatures above -50°F.

Resistance heating of the nitrogen is employed only during the cruise mode to reduce the consumption of gas. To keep the heater power consumption within practical limits they are designed for operation of the nozzles only at the 0.2 lb thrust level. Furthermore, the heaters will remain "ON" continuously during cruise because of the thermal time constant involved; pulsing the heaters just prior to actuating the valves would result in high peak powers to achieve the desired temperature. There are two (or four) gas feed tubes, corresponding to the two (or four) nozzles per assembly, entering the heater (Figure 5-24). These tubes have an inside diameter of 0.120 inch, a wall thickness of 0.013 inch, and are of 304 stainless steel. The two-wire heater lead entering the heater connects to the Balco heater element. The core, around which the heater element and gas feed tubes are wound, is a copper slug 0.5 inch in diameter and 2 inches long. The tubes and heater wire comprise the heat exchanger and are imbedded in insulation.

The solenoid valves and pressure regulators employ redundant seat design. This has been shown to keep leakage negligible, as well as provide the added reliability of continued operation in the presence of one valve seat (open) failure. Redundant seating adds little to component weight and is presently being used in other spacecraft developed at TRW.

The nitrogen storage tanks are approximately 19 inches in diameter and constructed of titanium with a 1.5 burst safety factor. The plumbing and manifolds are constructed of stainless steel and the joints welded, where applicable, to minimize system leakage. In cases where welded joints cannot be employed, conventional "B"-nut fittings will be used with metal gaskets.

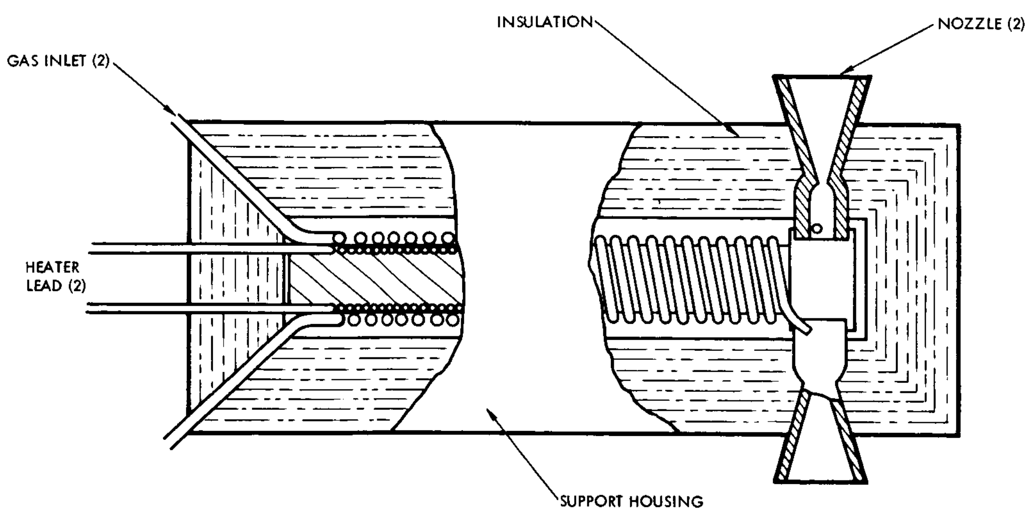


Figure 5-24
THE RCS GAS HEATER AND NOZZLE ASSEMBLY make use of a helical heater element and helical gas line.

Alternate approaches to reaction control that have been studied are contained in Appendix K.

5.3.7 Control Electronics Assembly

The guidance and control electronics assembly contains the amplifying switching and logic circuits which process output signals from the inertial reference unit, coarse and fine sun sensors and Canopus sensor. Its preliminary specification appears in Figure 5-25. The functional block diagram of the guidance and control electronics appears in Figure 5-26.

- Rate Nulling - For the reduction of planetary vehicle angular rates to 0.01 deg/sec about each axis, the rate-connected gyro signals are applied to the pitch, yaw and roll axis reaction control system valve drivers through the parallel DRI/lead-lag controller. The lead-lag portion of the controller operates on large error signals to produce rapid convergence and the DRI portion operates when the error is low.

PRELIMINARY SPECIFICATION

Guidance and Control Electronics Assembly

Purpose

Provides selection and amplification of input signals from celestial and inertial sensors. Provides rate damping by means of derived rate modulator circuits or lead lag networks. Provides output signals of suitable power level for operation of the RCS valves, TVC actuators and the C-1 engine propellant valves.

Performance Characteristics

SENSOR SIGNAL INPUTS

Canopus roll error signal	0-14 VDC
Canopus acquisition signal	0 or 5 VDC
Coarse sun sensor outputs	0 to 5 VDC
Fine sun sensor output	0-5 VDC
Pitch, roll yaw gyro outputs	0-15 VDC

OUTPUT SIGNALS

Pitch, roll, yaw RCS solenoid valves
No. 1 and No. 2 TVC actuators
Four, C-1 engine solenoid valves

Physical Characteristics

Size	7 x 6 x 11 in.
Weight	13 lb
Power	25 watts avg 38 watts peak

Figure 5-25

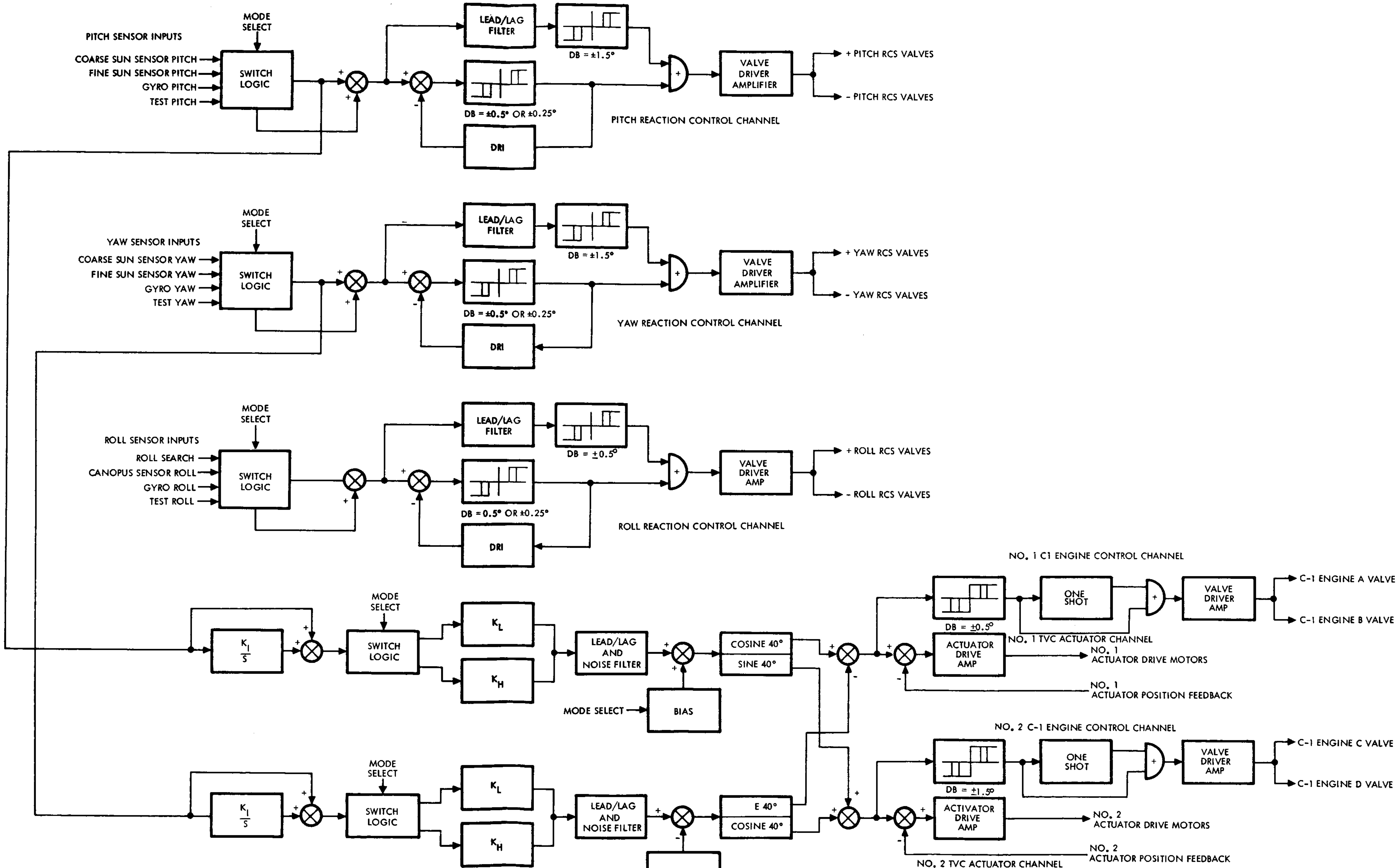


Figure 5-26

GUIDANCE AND CONTROL ELECTRONICS ASSEMBLY includes RCS and TVC channels. Redundancy and details of logic and mode switching are not shown to avoid complication of the diagram.

- Sun Acquisition - When the sun is sensed by the coarse sun sensors, their output signals are mixed with rate connected gyro signals and applied to the pitch and yaw axis control electronics. When the sun is acquired by the fine sun sensor, the fine sun acquisition signal causes the inputs of the pitch and yaw axis to receive the fine sun sensor signals, and the pitch and yaw gyro rate signals to be removed.

- Canopus Acquisition - For Canopus acquisition a 0.10/sec Canopus search signal is mixed with the roll rate gyro signal and is applied to the roll axis. When Canopus is acquired, the Canopus acquisition signal switches the roll axis input to the Canopus sensor error signal and removes the roll gyro rate signal.

- Cruise Mode - In the cruise mode, pitch and yaw positions of the planetary vehicle are determined by inputs from the fine sun sensor, with roll position determined by the Canopus sensor signal. During this mode, the gyros are not used and rate damping is obtained by means of the DRI network. The deadband select signal reduces the limit cycle amplitude to less than 0.25 degree during the photo-imaging mode.

- Maneuvers - For attitude maneuver, the celestial sensor input signals are removed from the pitch, yaw and roll axes and gyro position signals are applied. As individual gyros are torqued in accordance with turning rate commands, the respective axes will receive signals through the control electronics to accomplish the desired attitude turns.

- Inertial Attitude Hold - When inertial attitude hold is desired, the gyro position signal is applied to the RCS through the parallel DRI/lead-lag controller.

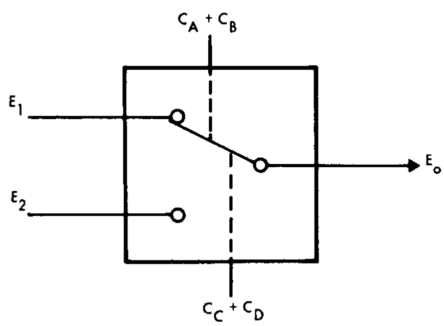
- Thrust Vector Control - During operation of the main or C-1 engines gyro position signals are applied to the pitch, yaw and roll axes through the pitch and yaw thrust vector control channels and the roll reaction control system channels. When using the main engine these signals pass through a selectable gain network set for high or low thrust, a lead-lag network for rate compensation and a coordinate transformation network to transform the control signals from the sensor axes to the actuator axes. They are then used to drive the engine gimbal actuators. A selectable bias signal is inserted ahead of the coordinate transformation to compensate for the predictable portion of the center of mass offset and so reduce the thrust vector directional errors.

When the C-1 engines are used the output of the transformation network is applied to an on-off, minimum off time controller to duty cycle modulate the flow of propellant to the engines. Thus a controllable torque about the vehicle pitch and yaw axes is produced to compensate for thrust mismatch, misalignment and center of mass offset.

For use with both the main engine and the C-1 engines roll control is obtained via the reaction control system.

5.3.7.1 Mode Control Switching Electronics

The basic circuits used in the mode switching portion of the electronics assembly are analog gates and summing amplifiers. The analog gate is described functionally in Figure 5-27. It is essentially a solid state latching single pole double throw switch. The analog gate that will be used is the microcircuit developed and manufactured by TRW on a previous program. The schematic appears in Figure 5-28. A field effect transistor is used



COMMAND	E_o
C_A	E_1
C_B	E_1
C_C	E_2
C_D	E_2

Figure 5-27
THE ANALOG GATE is essentially a solid-state latching single-pole double-throw switch.

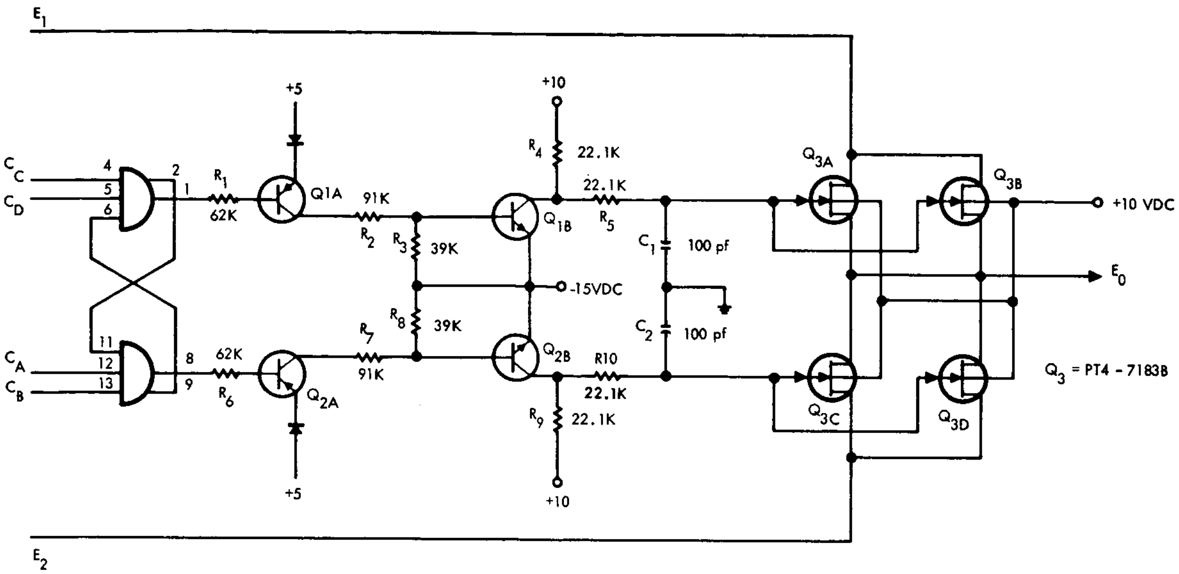


Figure 5-28
THE ANALOG GATE MICROCIRCUIT makes use of a field effect transistor as the switch that is driven by the level shift gate and flip-flop.

as the switch which is driven by level shift gate and flip-flop. To avoid catastrophic effect due to a single failure, six analog gates in a series-parallel configuration are used in place of one analog gate as shown in Figure 5-29.

The summing amplifier used is a triple redundant self-correcting amplifier developed by TRW. The configuration of this amplifier appears in Figure 5-30.

5.3.7.2 Controller and Valve Driver Electronics

This portion of the electronics assembly contains basically three identical channels of electronics. A single channel contains a level detector and switching amplifier with DRI feedback in parallel with a lead-lag network and another level detector amplifier combination and valve drivers. The single channel block diagram appears in Figure 5-31. Little discussion is necessary of the various blocks because of their straightforward nature. However, the method used to implement the redundancy deserves discussion. The system of redundancy to be utilized is the triple redundancy with majority voting. The voting will take place in the valve drivers. The implementation of the redundant system appears in Figure 5-32. Three non-redundant derived rate modulators with parallel lead-lag (as described in Figure 5-31) are connected in parallel. The bi-level outputs of the position negative level detectors are connected in

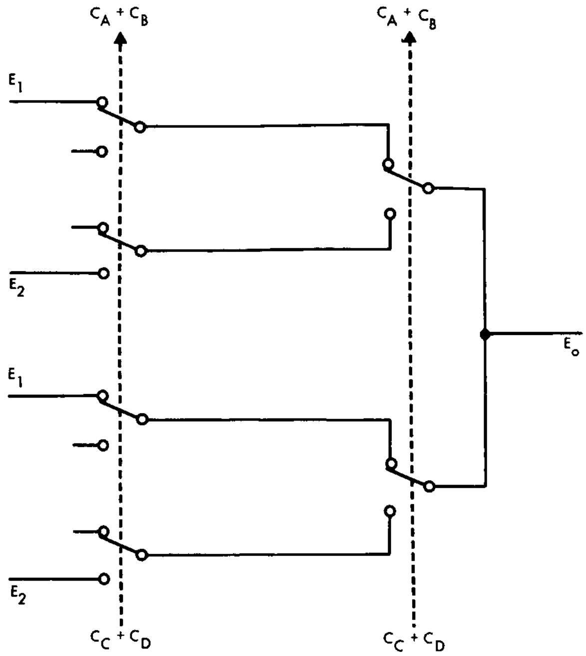


Figure 5-29
SIX ANALOG GATES in a series-parallel configuration protect against catastrophic failure with failure of one component.

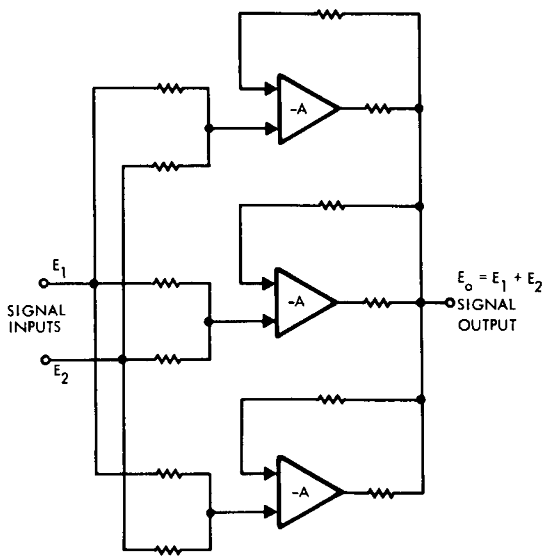


Figure 5-30
THREE OPERATIONAL AMPLIFIERS are interconnected to form a triply redundant summing amplifier in the guidance and control electronics assembly. The summing amplifier is thus unaffected by failure of any one component.

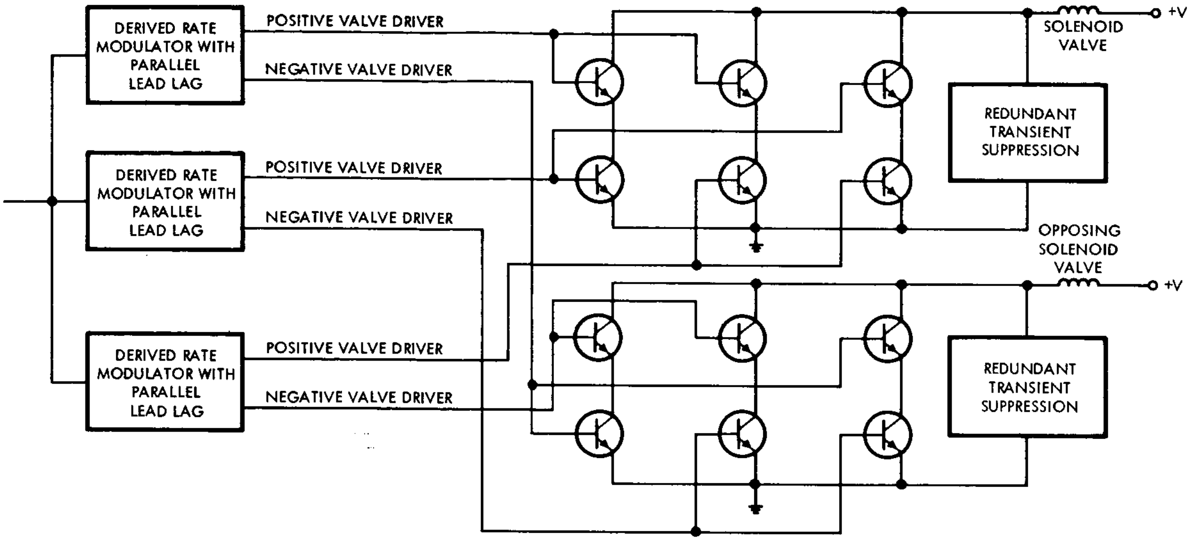


Figure 5-31
REDUNDANT IMPLEMENTATION OF SINGLE CHANNEL MODULATOR AND VALVE DRIVER (MVDE) makes use of triple redundancy with majority voting.

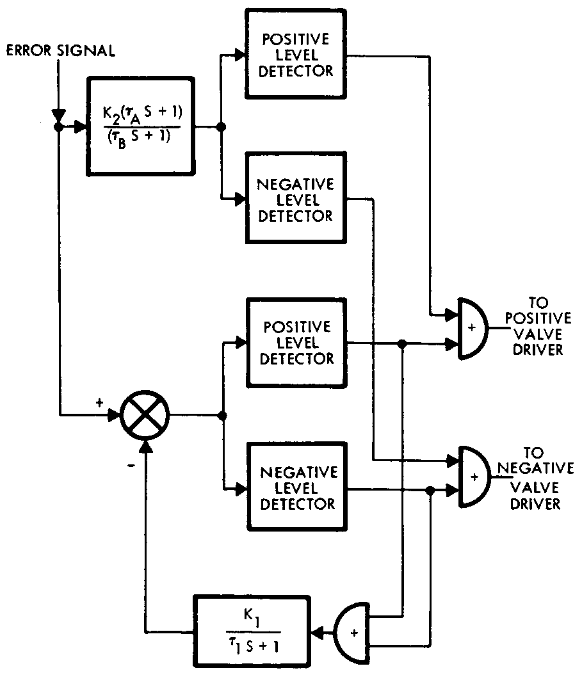


Figure 5-32
THE SINGLE CHANNEL NONREDUNDANT DERIVED RATE MODULATOR is equipped with parallel lead/lag feedback.

the majority voting valve drivers. The valve driver will actuate the solenoid valves only if at least two or more modulator outputs are present. Therefore if any single modulator failed, the valve-drivers will respond to the other two of the three modulators. The valve drivers are also redundant in that any single transistor can fail and the operation is not affected.

The thrust vector control channels are implemented with a triple redundant voting scheme. Each of the two channels in the thrust vector control electronics is mechanized to drive an actuator containing three DC motors. The position transducer is a digital encoder. There are three encoders per channel. The proposed redundant scheme was used successfully on the LMDE throttle control. The block diagram appears in Figure 5-33. The fail safe power amplifier is similar to the amplifiers used in the LMDE throttle controller. The schematic appears in Figure 5-34.

The mechanization selected for the thrust vector control actuators is a DC torque motor/ball screw combination; the block diagram and Preliminary Specifications for the thrust vector control actuator is shown in Figure 5-35. As will be discussed in Appendix R, the DC motor/ball screw approach is preferred on the basis of reliability and the backlog of previous spacecraft experience.

5.3.8 Thrust Vector Control Actuator

The thrust vector control actuator is used to gimbal the engine nozzle through commands received from the guidance and control subsystem. A command voltage produces a current in the armature of the DC motor. This current produces a torque which rotates the armature and the ball screw nut (Figure 5-36). The ball screw nut converts the rotary motion of the armature to a linear motion of the actuator output shaft which causes the engine nozzle to rotate about its gimbal in one axis. Two thrust vector control actuators placed at 90 degrees on the engine nozzle, therefore, result in the 6 degree half-angle cone required for thrust vector pointing. Velocity feedback within the actuator is accomplished by means of the back EMF generated by the DC motors and position feeback is obtained by three rotary digital encoders

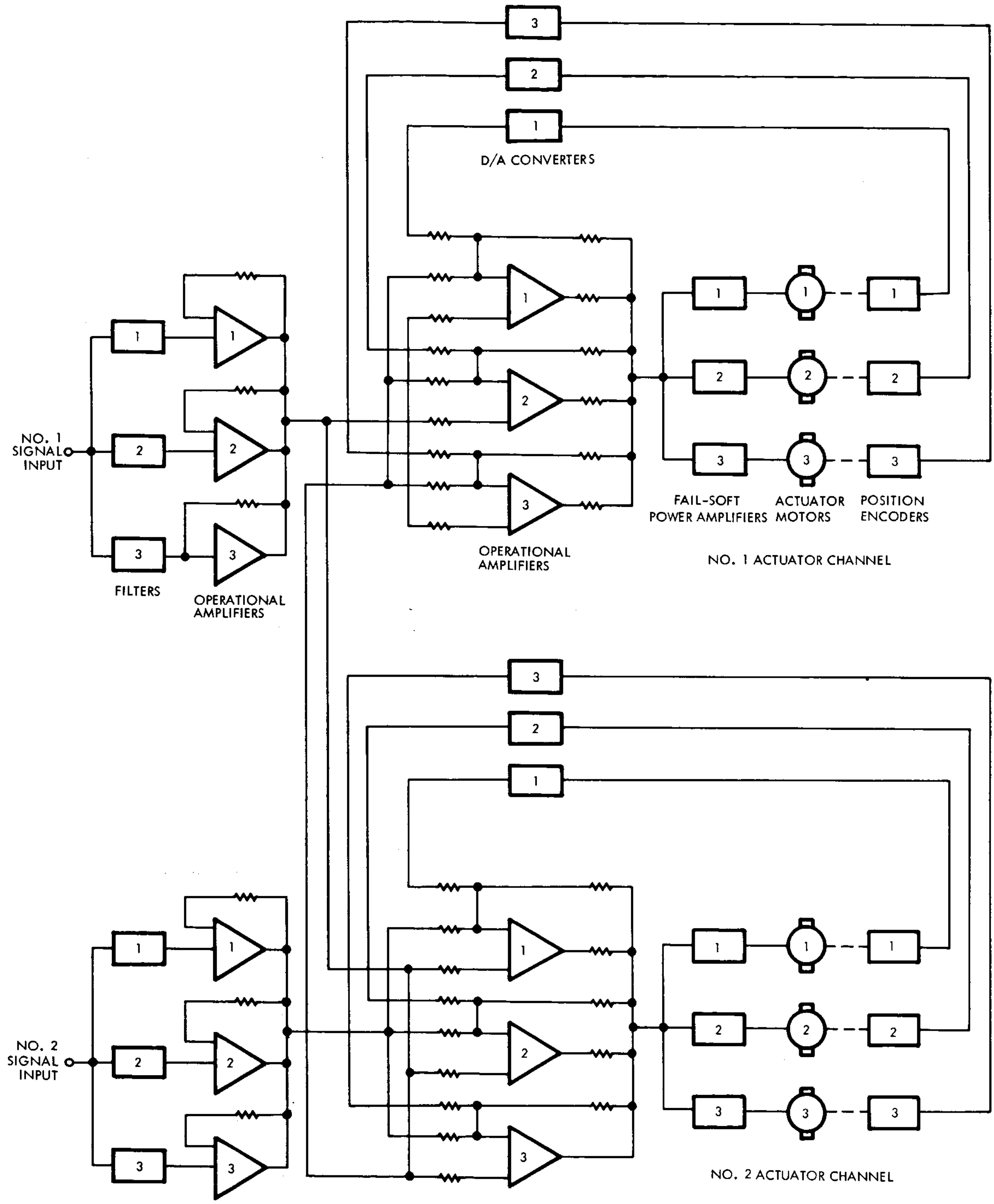


Figure 5-33
THE TVC ELECTRONICS PORTION OF GUIDANCE AND CONTROL ELECTRONICS ASSEMBLY makes use of extensive redundancy.

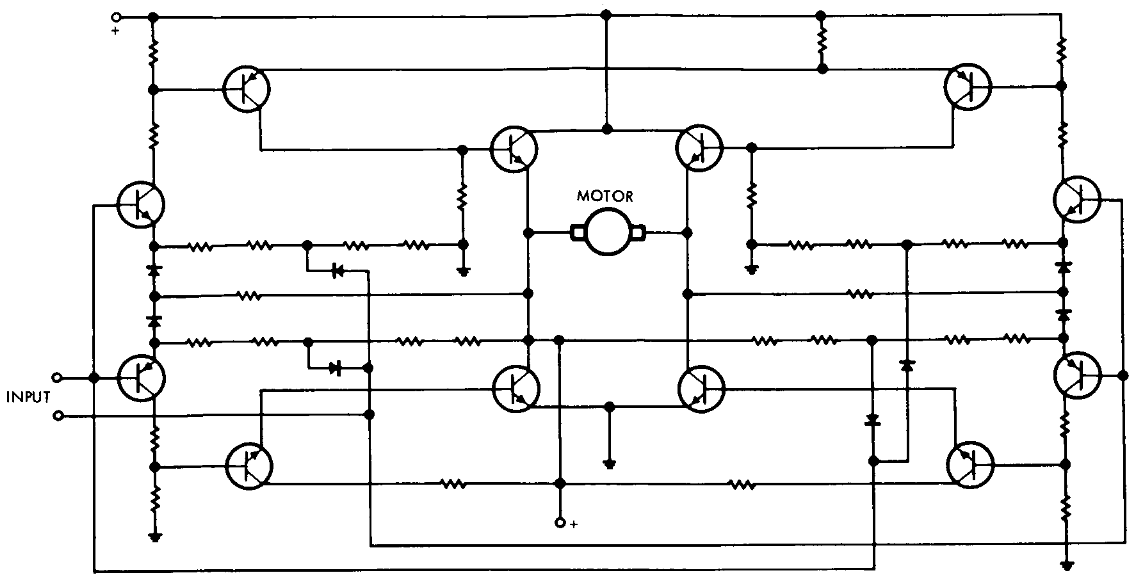


Figure 5-34
FAIL-SAFE POWER AMPLIFIER is similar to the amplifiers used in the LMDE throttle controller.

PRELIMINARY SPECIFICATION

TVC Actuator

Purpose

Provides linear force output for gimballing main engine during midcourse trajectory corrections, orbit insertion and orbit trim maneuvers. One actuator each used in the two gimbal axes.

Performance Characteristics

Actuator stroke	4.1 in.
Force output	225 lb
Maximum slew rate	1.67 in./sec
Extended length	18 in.
Maximum acceleration	16.7 in./sec^2
Maximum de-energized holding force	50 lb

Physical Characteristics

Overall length	18 in.
Maximum diameter	7 in.
Weight	24 lb
Heretically sealed	
Power (DC)	130 watts avg 189 watts peak

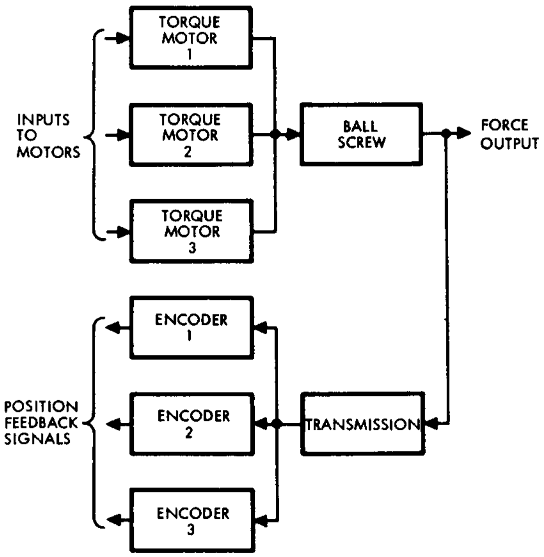


Figure 5-35

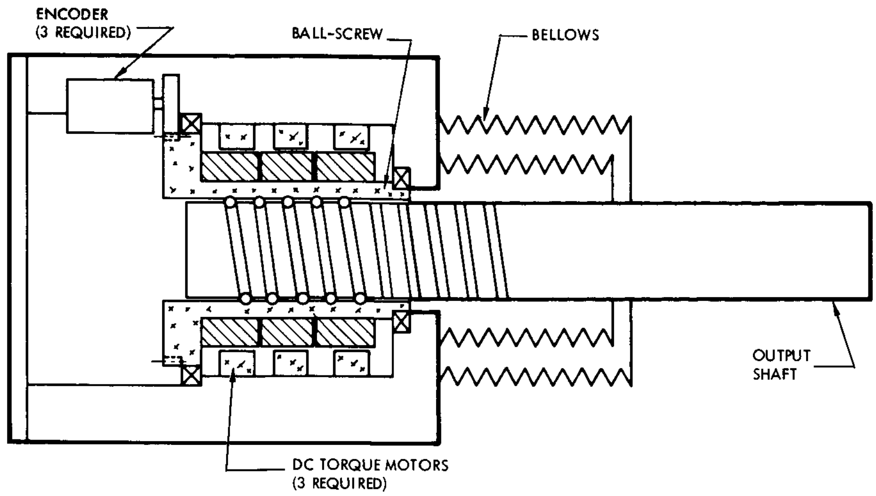


Figure 5-36
MECHANICAL SCHEMATIC DIAGRAM OF TVC ACTUATOR showing the arrangement of working parts and the use of three motors operating in parallel for failure protection.

geared to the ball nut. Three position encoders are provided to supply three independent feedback signals to the majority voting redundant electronics. In addition, three motors in tandem are provided for redundancy; any two motors are capable of driving the load.

In order to size the DC motor required for the thrust vector control actuators, the loads carried by the actuator must be known:

Engine thrust offset torque:	195 ft-lb
Engine gimbal friction:	67 ft-lb
Flexline torque (max):	66 ft-lb
Maximum engine steady-state load:	328 ft-lb

Assuming a ball-screw lead of 0.50 inch and a ball-screw pitch diameter of 2.0 inches, the mechanical advantage between the motor and the load is approximately 225:1. The total load at the motor can be found by adding the steady-state engine load, the motor brush friction, and the inertia loads reflected to the motor:

Steady-state engine load

$$\frac{(328 \text{ ft-lb})}{225} = 1.45 \text{ ft-lb}$$

Actuator friction (ball-screw, brushes, encoder)

$$(3)(0.035 \text{ ft-lb}) = 0.105 \text{ ft-lb}$$

Inertia loading (reflected to motor)

a) Engine nozzle inertia

$$\frac{(50 \text{ slug-ft}^2)}{225^2} = 0.00099 \text{ slug-ft}^2$$

b) Actuator inertia (motor, ball-screw, encoder)

$$(3)(0.00076 \text{ slug-ft}^2) = 0.00228 \text{ slug-ft}^2$$

c) Total inertia torque

$$\frac{(225)(0.00228 + 0.00099 \text{ slug-ft}^2)(50^\circ/\text{sec}^2)}{57.3^\circ/\text{rad}} = 0.643 \text{ ft-lb}$$

Total motor load: 2.198 ft-lb

This peak torque can be handled by two DC torque motors such as the Inland T-4036, which has a peak rated output of 1.8 ft/lb. Several small motors rather than one large motor are preferred in this application in order that the diameter of the packaged assembly be relatively small. As mentioned previously, three motors are provided in tandem for redundancy purposes.

The proposed configuration for the thrust vector control actuators is shown in Figure 5-35. It is noted that the entire drive assembly is contained within a hermetic sealed housing for protection against the environment. The output shaft will be sealed with two concentric welded metal bellows.

The performance and physical characteristics for the thrust vector control actuators are shown in Figure 5-35.

The peak power required for the motor is 375 watts for two axes. This peak occurs when the control system is commanded by a step. Since steady-state loads make up most of the motor load, the average power is lower and is estimated at 260 watts for two axes.

The total weight of two thrust vector control actuators is as follows:

Motors (redundant), housing:	38 lb
Ball screw	4 lb
RFI filters, encoders, etc.	6 lb
Total Weight	48 lb

The length of the actuator has been specified by the requirements and includes the distance from the actuator mount to the engine attach point.

6. COMPUTER AND SEQUENCER SUBSYSTEM

6.1 SUMMARY

The computer and sequencer subsystem provdes discrete commands and serial data for the control of spacecraft operations and scientific instruments throughout the Voyager mission. It consists of a special-purpose sequencer, counters for velocity increment measurement, a function generator for planetary scan platform and antenna pointing, and peripheral units that form interfaces with other subsystems. The commands and control data for a complete mission will be stored in the sequencer before launch. The subsystem performs sequencing on a mission independent basis, using sensor outputs for control or generation of stored command sequences. Modification of the stored program commands is required to correct for trajectory dispersions, mission changes, or detected failures. In the last case, all stored program command parameters and sequences can be modified in flight by ground commands.

Preliminary specifications for the computer and sequencer, appear in Figure 6-1. Its outline in the spacecraft appears in Figure 6-2.

To enhance the reliability of the recommended Voyager subsystem, a complete backup sequencer with a full range of stored program commands and sequences is provided. In addition, many stored program command functions may be carried out directly in response to ground commands. The computer and sequencer subsystem sends its commands directly to spacecraft subsystems and does not use the command subsystem decoder.

A summary of the command sequences carried out by the computer and sequencer during the successive phases of a nominal mission is given in Table 6-1.

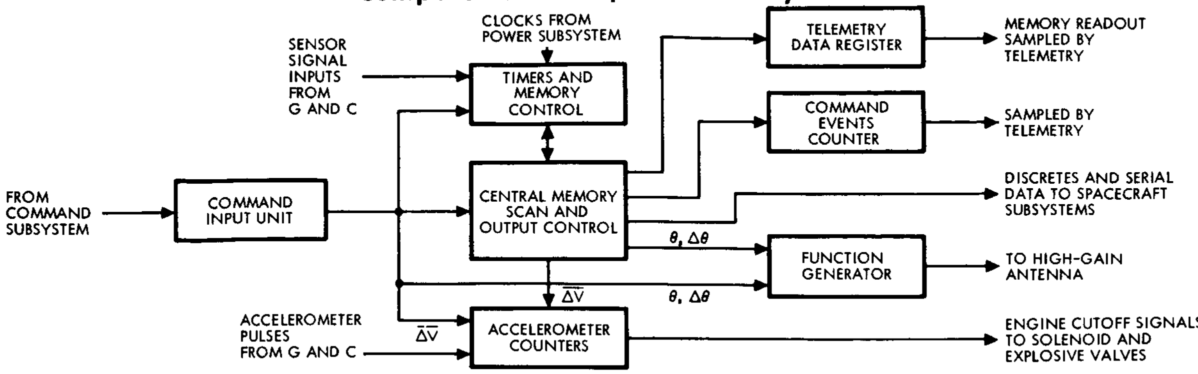
6.2 SUBSYSTEM DESIGN AND PERFORMANCE

6.2.1 Organization and Operation

A functional block diagram of the primary sequencer is shown in Figure 6-3. The command input section accepts messages from the command subsystem, processes the data, and feeds it to the appropriate section within the computer and sequencer. The ground command data enters the

PRELIMINARY SPECIFICATION

Computer and Sequencer Subsystem



Performance Characteristics

Output voltage:	10 volts
Output current:	25 MA
Pulse width:	50 MS

Physical Characteristics

	PRIMARY	BACKUP
Weight:	18 lb	16 lb
Power:	20 watts	18 watts
Volume:	460 in.^{3}	400 in.^{3}
Reliability:	0.966	
Memory size:	512 words	512 words
Word size:	20 bits	20 bits

SEQUENCING CAPABILITY

- Up to 4 simultaneous sequences in operation
- Capacity for 32 sequences
- Sequences may be recycled from other sequences
- Capacity for storage of command sequences for entire mission
- Sequences alterable by ground command
- Sequences synchronized to variety of physical events
- Sequences may be enabled or inhibited by ground command

Figure 6-1

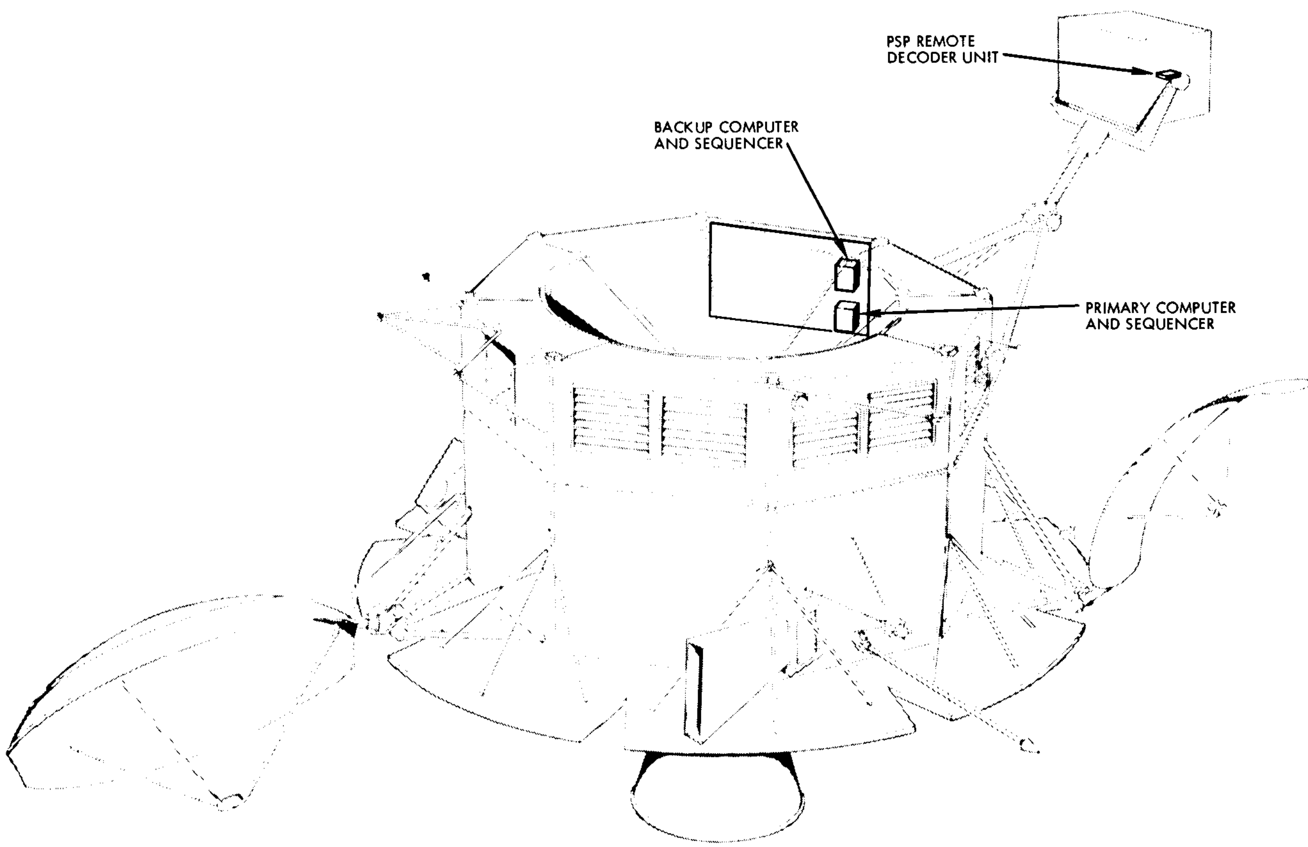


Figure 6-2
THE COMPUTER AND SEQUENCER SUBSYSTEM includes redundant main units and a remote decoder unit on the planetary scan platform.

Table 6-1. Computer and Sequencer Programs*

Post Separation Sequence (Maneuver Timer)	
Enable G and C and switch G and C to rate nulling mode	Initiated by booster/ spacecraft separation
Radio uplink to maximum coverage reception	
Radio downlink to low power low-gain antenna	
Pyrotechnic bus armed	
Mechanical hold-down mechanisms released	
Boom mounted low gain antennas deployed	
High gain antenaa deployed	
Medium gain antenna deployed	
PSP deployed cruise condition	
High gain antenna pointed to earth	
Radio downlink to low power high gain antenna	MOS verification of spacescraft cruise configuration
Prepare for celestial reference acquisition	
Acquisition Sequence (Maneuver Timer)	Initiated by previous sequence
Data storage and telemetry (DST) to store engineering data	
Radio downlink to low power low gain antenna	
Initiate G and C sun acquisition	
Prepare for Canopus acquisition	
Initiate Canopus acquisition	Enable timer on sun acquisition signal
Prepare for cruise operations	
Interplanetary Cruise Operations (Mission Timer)	
G and C to cruise operation (course limit cycle)	Initiated by previous sequence
DST to dump stored data mode	If required
DST to real-time engineering data mode	
High gain antenna repositioned	As required
Canopus sensor updated as required	
Radio downlink to high power low gain antenna	

*Table 6-1 presents a summary of the preliminary computer and sequencer subsystem command sequences for all mission phases. Included are the signals that initiate each sequence.

Table 6-1. Computer and Sequencer Programs (Continued)

<u>Maneuver Orientation and Velocity Adjustment (Maneuver Timer)</u>	Arrival date separation and midcourse trajectory corrections
High gain antenna repositioned to be earth-pointing after reorientation	
DST to real-time engineering data	MOS verification
DST to store engineering data	
Initiate G and C attitude orientation maneuver	
G and C to attitude hold (limit cycle) following maneuver)	
Radio downlink to low power high gain entenna	
DST to dump stored data	MOS verification
Initiate engine operation sequence (if enabled by MOS)	
Propulsion system pressurized	
Radio downlink to high power low gain antenna	
Pyrotechnic bus armed	
DST to store engineering data mode	
GC to engine operation control	
Initiate engine low thrust operation	Timed by accelerometer counter
G and C to attitude hold mode (following propulsion operation)	
DST to dump stored data mode	
High gain antenna repositioned to earth-pointing after celestial acquisition	Preparation for reacquisition
DST to store engineering data mode	
Initiate acquisition sequence	
<u>Maneuver Orientation and Velocity Adjustment (Maneuver Timer)</u>	Orbit insertion
High gain antenna reposition to be earth-pointing after reorientation	
DST to real-time engineering data	MOS verification
DST to store engineering data	
Initiate G and C attitude orientation maneuver	Timed by accelerometer counter
G and C to attitude hold (limit cycle)	
Radio downlink to high power high gain antenna	
DST to dump stored data	MOS verification
Initiate engine operations (if enabled by MOS)	

Table 6-1. Computer and Sequencer Programs (Continued)

Maneuver Orientation and Velocity Adjustment (Maneuver Timer) (Continued)	
Pyrotechnic bus armed	
DST to store engineering data	
G and C to engine operation control	
Initiate engine high thrust operation	Timed by accelerometer counter
G and C to attitude hold (following engine operation)	
DST to dump stored data	
Radio downlink to high power low gain antenna	
High gain antenna repositions for celestial reacquisition	
DST to store engineering data	
Initiate acquisition sequence	
Post Orbital Insertion Operation (Maneuver Timer)	
G and C to limit cycle	
Radio uplink to maximum coverage reception	
Radio downlink to high power high gain antenna	
Medium gain antenna repositioned to earth-pointing	If required
DST to dump stored data	
DST to real-time engineering	
PSP to automatic Mars tracking	
G and C to fine limit cycle	
Orbital science preparations and calibration	Orbit and science timers optional
DST to real-time science and store video data	
DST to dump stored video and store science data	
DST to real-time science data	MOS verification and science sequence update
Orbital science operation initiated	Terminator synchronization optional
DST to real-time science and store video data	
DST to dump stored video data and store science data	
DST to real-time science data	

Table 6-1. Computer and Sequencer Programs (Continued)

Spacecraft-Capsule Separation (Maneuver Timer)	
Preparation for separation	Maneuver parameters updated
Radio uplink to maximum coverage reception	
Radio downlink to high power high gain antenna	
DST to capsule real-time data	
Orbital science operations initiated	Orbit and science timers optional
Capsule lid jettisoned	
DST to real-time capsule engineering and science data	
Capsule separation sequence initiated	
Capsule systems activated	
Capsule separation	
Post Separation Operations	
Capsule radio link to high rate data	
DST to real-time capsule engineering and store capsule video data	
DST to dump store data	Capsule out of line of sight
Cycle steps 2 and 3	Capsule within line of sight
Deploy PSP to orbital position	
Initiate orbital sequence	
Orbital Operation (noneclipse)	
Radio uplink to maximum coverage reception	
Radio downlink to high power high gain antenna	
DST dump stored data	
Initiate science sequence No. 1	Orbit and science timers synchronization to dawn terminator
Science sensors on	
DST to real-time science	
G and C to fine limit cycle	
PSP to automatic tracking	
Sensors to continuous data mode	
DST to real-time science and store video data	
Photo-imaging to data take	
Science sensors to intermittent data mode	
Science sensors to continuous data mode	
Photo-imaging OFF	
DST to real-time science	

Table 6-1. Computer and Sequencer Programs (Continued)

Orbital Operation (noneclipse) (Continued)	
Initiate science sequence No. 2	Synchronized to evening terminator
Sensors to intermittent data mode	
DST to real-time science and dump stored data	
Repeat sequence	Reinitiate orbit timer
Orbital Operations (eclipse) Orbit Timer	
G and C to inertial hold mode	Spacecraft enter eclipse
Sensors to continuous data mode	
G and C to coarse limit cycle	
Sensors to intermittent data mode	
Repeat sequence	

input unit as a pulse train having 20 data bits, 9 address bits to identify storage location, 2 data-class bits to identify message type, and 2 parity bits. Messages providing stored program commands include command times and sequence codes. Other messages to be stored in memory contain numerical data. The data-class bits identify the unit that receives the incoming message, i.e., memory, timer, accelerometer counter, or function generator. These bits also identify the "command enable" word that sets the enable bit in a memory word or timer. The last bit enables a command to be issued.

The four computer and sequencer timers each contain a sequence code and accumulated time when in operation. The contents of the computer and sequencer memory are scanned serially once each second and compared for sequence code and time coincidence simultaneously with the four timers. If both time and sequence coincidence with any timer occurs, and the command word enable bit is set, either a command discrete or a serial message is issued, depending on the function of that memory location.

A sequence can be synchronized to a particular event by setting the enable bit at the time of the event. This is accomplished by a signal from the terminator, limb-crossing or fine sun sensors, by a separation signal, by a Canopus acquisition signal, or by ground command. Setting the enable bit permits the controlling timer to begin timed pulse accumulation.

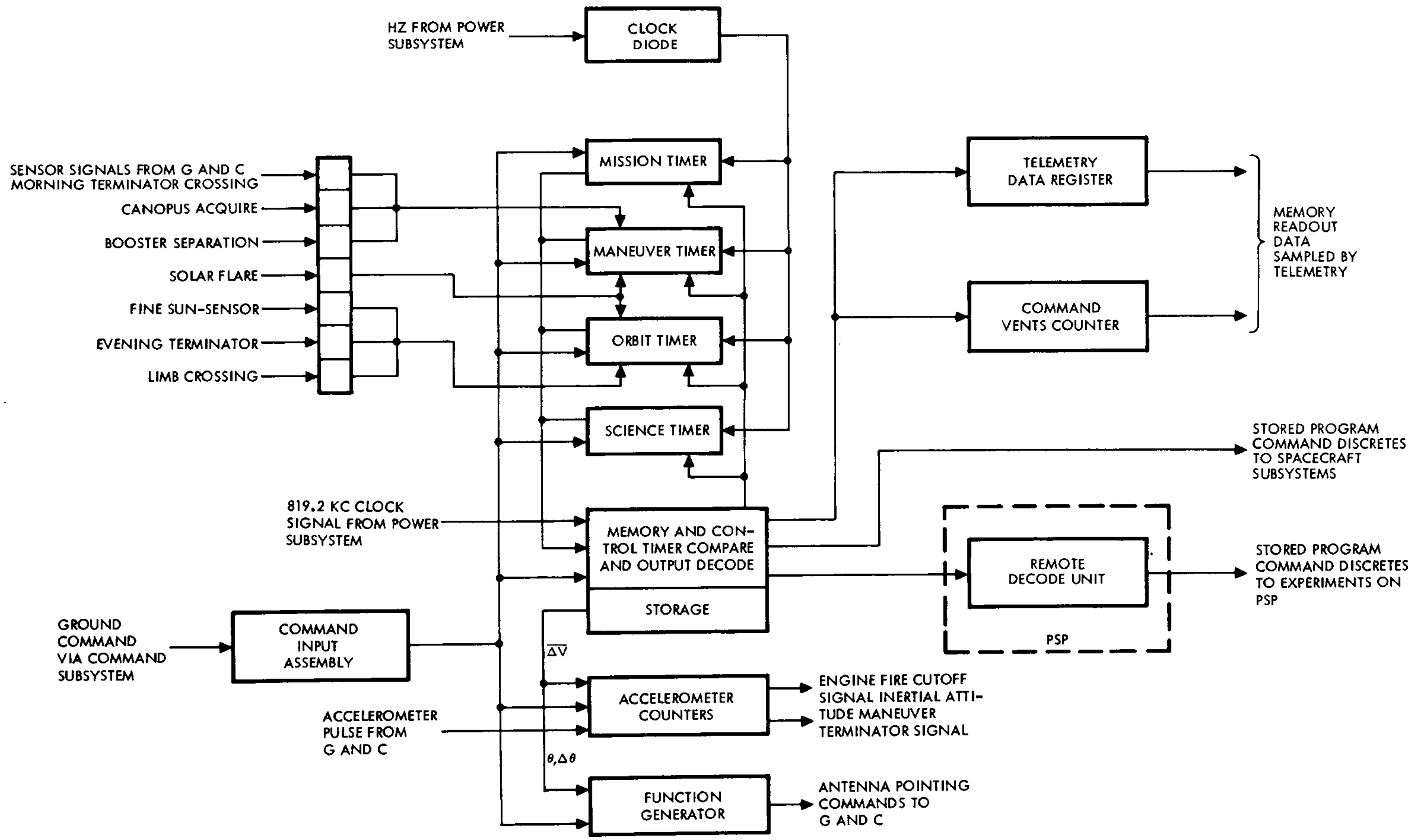


Figure 6-3
THE PRIMARY SEQUENCER OF THE COMPUTER AND SEQUENCER SUBSYSTEM: indicated is the flow of data among the assemblies. The remote decode unit contains the portion of the output decode matrix generating commands for scientific instruments and is located on the PSP. It operates as though it were an integral part of the sequencer.

The four timers can sequence commands concurrently. The mission timer has a granularity of one second, a range of 25 months and is used for sequencing long term events, beginning other sequences, and time tagging recorded data. Memory words are compared with the 14 most significant bits of this timer, thereby providing a 1.14 hour sequence granularity. The orbit timer has a four-second granularity and 18.2 hour range, and is used for all orbit science sequences which require the terminator crossing signal. The maneuver and science timers, with one second granularity and 4.5 hour range, are used for both maneuver and science sequences. The timers are capable of being loaded by stored program commands or ground commands via the command input unit.

The times or order of occurrence of stored program commands within any sequence can be modified or updated at any time by ground commands. The computer and sequencer can repeat an orbit sequence in a periodic fashion, e.g., measured from the morning terminator. It can also perform a series of orbit or science sequences initiated by the relatively long-term orbital time reference such that the initiation of each sequence in a string is in a fixed relation to the terminator or limb crossings. This permits the sequence to prepare for events to occur in the next orbit sequence. The mission timer is used to sequence long term events both from the memory and from the function generator.

The function generator is used to generate antenna pointing commands. No sequence designation is required for the function generator and the data output time is established by sampling the mission timer. Computations are performed from initial data obtained either from stored program commands or from ground command.

Engine cutoff commands are issued from the accelerometer counters. The accelerometer counters accumulate accelerometer pulses during engine burn and issue a cutoff signal when the count reaches the required incremental velocity correction. The cutoff point is established by preloading the counter with the complement of the incremental velocity and letting the overflow pulse cut off the engine. The overflow pulse is sent to the correct cutoff valve by first enabling the appropriate one of five gates, one for each of five sets of valves.

The accelerometer counters also provide cutoff commands for inertial attitude maneuvers. For this mode of operation, the counter is preset with the complement of the maneuver time. From initiation of the maneuver, by stored program command, the counter accumulates pulses at a constant 10 pps rate until overflow occurs. This signal is fed to the guidance and control subsystem to terminate the maneuver.

Finally, two telemetry registers provide indications of computer and sequencer status to the ground. One register provides stored data. Memory words are loaded into the register and are sampled by telemetry sequentially. Gating logic prevents interference of the loading and sampling operations. The other register is used to count the number of stored program commands issued since prior interrogation by telemetry.

6.2.2 Redundancy and Failure Modes - Subassembly Redundancy

Cooperative multichannel redundancy is provided in the sequencing of computer and sequencer stored program commands. This is accomplished by permitting maneuver sequence to be controlled from more than one timer. The maneuver, orbit and science timers are needed to control concurrent orbital operations synchronized separately to terminator and limb crossing points, but they are also capable of providing some backup capability for each other. If one fails, the others can be loaded from the ground to initiate sequences and time events.

Functional redundancy is also provided by ground backup loading capability. Each timer can be separately addressed and loaded from the command subsystem. Individual sequences can be initiated from the ground and controlled by an alternate timer. In the same manner, ground command backup is provided for sequence synchronization.

Similar redundancy is provided for the accelerometer counters and function generator. Normally these units are fed from the sequencer memory, which contains the initial quantities and start commands. However, as a backup, they can also be loaded and operated from the ground in the event that the memory fails.

If either of the accelerometer counters fail, the engine can still be shut down by using a stored program command controlled from a sequence timer. Stored start and stop commands are provided for each of the engine operating modes. The stop commands are issued over output lines to the same valves controlled from the accelerometer counter and can activate those valves if the counter fails. Should the primary engine fail to start during the orbit insertion mode, this failure is detected by the accelerometer and the backup C-1 engine insertion mode is initiated. The C-1 engine insertion mode is initiated. The C-1 engines are started and the stored pitch and yaw angular rate program is initiated. This program is required because, with the low thrust and long burning time, a constant attitude firing would result in an orbit which would impact Mars. The desired constant pitch and yaw rates are approximated by torquing the gyros at the standard rate of 0.2 degree/sec with an appropriate duty cycle.

Should the Mars limb or terminator crossing detector sensor signal input control fail to operate, sequencing initiate times may be controlled by properly timing ground commands. Orbit science sequencing can thus be synchronized to Mars lighting conditions even if the sensors themselves fail to operate.

In addition to cooperative multichannel or functional redundancy, direct commands via the command subsystem provide backup or alternate means of accomplishing many computer and sequencer functions. Direct command backup is not appropriate, however, for all functions. Because of the danger in using direct commands to control the attitude maneuvers and propulsion start-stop, an alternate means for on-board control is provided. This is accomplished by the backup sequencer. Stored program command capability from the central memory of this unit duplicates that of the primary sequencer.

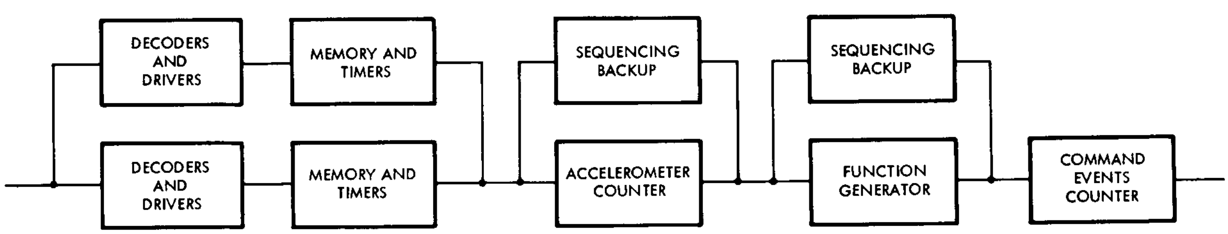
Antenna pointing commands are also available in a reduced capability. These commands are stored in memory and can be updated by ground command as desired. These commands point both the medium and high-gain antenna hinge angles, and the high gain antenna shaft angle.

Updating of stored parameters in the backup sequencer is performed separately from the primary. The backup sequencer is normally in a standby mode with no power to memory and the majority of the logic. Only the backup mission timer is kept running and synchronized with the primary mission timer. Power to the major portion of the backup sequencer is controlled by ground command via the command subsystem or automatically applied if certain specific failures occur in the primary. A loss of the internal clock is one such example.

The backup sequencer provides the capability for memory verification via telemetry in the same manner as the primary.

The reliability block diagram for the computing and sequencing unit is shown in Figure 6-4. The figure shows that the major components in the computer and sequencer are redundant. However, in this unit, failure of a component in the on-line equipment will necessitate complete switchover to the standby sequencer. Failure of the accelerometer counter is not a system failure as the sequencer issues backup engine turn-off commands in the event the accelerometer or the accelerometer counter fails. Also, the failure of the function generator will not result in system failure as the sequencer also can issue backup antenna pointing commands to the high gain and medium gain antennas. Antenna pointing can then be accomplished by either employing the function generator or the computing sequencing backup. Finally, the command events register simply counts the commands issued, and failure of this equipment will mean that the ground control will not have a record of this information. This register is then effectively a piece of test or verification equipment and the failure could not be construed as a system failure.

The figure lists the computing and sequencing unit equipment, the component failure rates, and the component probability of success for the



NAME	FAILURE RATE (λ) (BITS PER 10^9)	PROBABILITY OF SUCCESS $e^{-\lambda T}$ T= 6800 HR
DECODER AND DRIVERS	10,890	0.929
MEMORY AND TIMERS	18,965	0.879
ACCELEROMETER COUNTER	1200	0.992
COUNTER BACK-UP	500	0.997
FUNCTION GENERATOR	1500	0.990
ANTENNA POINTING BACK-UP	500	0.997
COMMAND EVENTS REGISTER	570	0.996

Figure 6-4

RELIABILITY BLOCK DIAGRAM OF THE COMPUTER AND SEQUENCER SUBSYSTEM: the figure lists the computer and sequencer subsystem equipment, the component failure rates, and the component probability of success for the 6800 hour mission. The reliability for the computer and sequencer subsystem is 0.981.

6800 hour mission. The reliability expression for the computing and sequencing unit Is:

$$R_{Unit} = \{R_1R_2\ [1 + (\lambda_1 + \lambda_2)T]\}\ [R_3R_4\ (2-R_3R_4)]\ [R_5R_6\ (2-R_5R_6)]$$
$$= 0.981$$

6.2.3 Input/Output List

A summary of input/output requirements for the computer and sequencer is given in Table 6-2. Shown in the table are the commands which the computer and sequencer is required to perform and the input signal it receives from other subsystems.

6.3 COMPONENT DESCRIPTION

6.3.1 Mission Timer

The mission timer (Figure 6-5) is used for sequencing long term events, initiating other sequences, and time tagging science data. The timer is incremented at 1-second intervals and has a range of 2^{26} providing a full-cycle period of 25 months, 16 days and 7.35 hours. Memory words are compared with the 14 most significant bits thereby providing a 1.14 hour granularity.

The mission timer is separated into two sections. The first section (lower half) contains the 12 least significant bits. This section is incremented at 1-second intervals. Upon completion of a full cycle, a pulse is fed to the second section (upper half). This section contains the most significant 14 bits of the timer and a fixed mission time sequence code. Memory words are compared with this section only.

The mission timer may be loaded from the computer and sequencer memory or directly from the command subsystem via the command input unit. Each section of the mission timer is loaded individually. Counting in the timer is inhibited until an enable bit is set. This may be set by discrete command from either the computer and sequencer or command subsystem, or directly by operations support equipment prior to launch. Provided the enable bit is set, the timer is incremented by cycling the contents of the two sections. The upper half section is incremented during the second word time simultaneously with memory compare.

Table 6-2. Summary of Input and Output Requirements

Subsystem	Function	No. of Commands	Remarks
Command Outputs			
Guidance and Control	Inertial hold mode	2	All AXES inertial
	Enable sun acquisition	1	
	Canopus sensor on/off	2	
	Canopus sensor select	2	
	TVC actuator select	2	
	Gyro power on/off	2	
	Accelerator enable/disable	2	Used in propulsion operation
	TVC low/high gain select	2	
	TVC on/off	2	
	TVC test	1	
	Fine/coarse control select	2	Deadband
	Thrust reaction control	2	High/low select
	Roll-spin Canopus search	2	
	Star acquisition gate override	2	
	Roll incremental maneuver	2	
	Roll axis inhibit	2	
	High gain antenna hinge axis	1	Serial (quantitative) command
	High gain antenna shaft axis	1	Serial (quantitative) command
	Medium gain antenna hinge axis	1	Serial (quantitative) command
	Enable terminator-limb crossing detectors	2	
	Roll axis override	2	
	Enable maneuver	2	
	Canopus sensor update	1	Pulse per update
	Acquisition	2	Initiate automatic search
	Gyro reference assembly test	1	
	Accelerometer test	1	
	Set turn polarity	1	
	Pitch start/stop	2	
	Roll start/stop	2	
	C-1 engine enable	1	
	C-1 engine on/off	2	
C and S (internal)	Maneuver sequence select	32	Load timers
	Science sequence select	6	Load timers (orbit sequences)
	Load accelerometer CNTR A	4	
	Load accelerometer CNTR B	4	
	Load function generator	6	
	Load mission timer L. H.	1	
	Load mission timer U. H.	1	
	Enable accelerometer outputs	12	Engine cutoff and inertial maneuver termination command enables
	Emergency acquisition sequence select	1	Conditional command followed by serial data for timer
Communications	Transmitter/antenna	3	Power/gain selection
	Exciter-power Amp No. 1 on/off	2	
	Exciter-power Amp No. 2 on/off	2	
	Low power transmitter on/off	2	
	Receiver maximum gain high gain antenna	1	
	Receiver maximum gain medium gain antenna	1	
	Receiver maximum coverage	1	Special mode in the event of loss of attitude control
	Low data rate capsule relay receiver on/off	2	
	High data rate capsule relay receiver on/off	2	
	Capsule relay link power on/off	2	
Telemetry and Data Storage	Data modes	6	
	Data rates	3	
	Store high rate capsule data	1	
	Store maneuver data	1	
	Recorder playback start/stop	12	
	Recorder record start/stop	12	
	Recorder speed select A/B	2	
Capsule	Start/reset capsule	2	
	Capsule power on/off	2	
	Open canister	1	
	Capsule internal/external power	2	
	Release capsule	1	
	Capsule thermal control on/off	2	
Science	PSP experiment control	144	These commands assigned as a block located on the PSP
Electrical distribution	Selection of solenoid Valves and explosive Valves	13	Starting and stopping orbit Insertion propulsion/midcourse Maneuvers
	Depressurize helium and fuel tanks	1	
	Depressurize oxidizer tank	1	
	Release low gain antenna	1	
	Release high gain antenna	1	
	Release medium gain antenna	1	
	Release PSP	2	Intermediate and full position
	Uncage gimbal PSP	1	
	Ordnance safe	1	
	Ordnance pre-arm	1	
	Arm propulsion	2	Midcourse and orbit insertion
	Arm propulsion depressurization	1	
	Arm antenna release	1	
	Arm PSP release and uncaging	1	
	Open C-1 engine fuel and oxidizer isolation valves	1	

Subsystem	Type	Source
Inputs	Dawn terminator crossing signal	G and CS
	Evening terminator crossing signal	G and CS
	Limb crossing signal	G and CS
	Solar flare signal	Science
	Booster/spacecraft separation signal	Adapter
	Fine sun sensor signal	G and CS
	Canopus acquired signal	G and CS
	Accelerometer pulses	G and CS
	Input commands	Command subsystem
Clocks and Power	815.2 kHz	Power subsystem
	400 Hz	Power subsystem
	8 pps	Command subsystem data input rate
	50 vdc	Unregulated primary power from power subsystem

*A summary of the computer and sequencer subsystem input and output requirements is shown in Table 6-2. Included is a preliminary stored program command list grouped by subsystem which receive them.

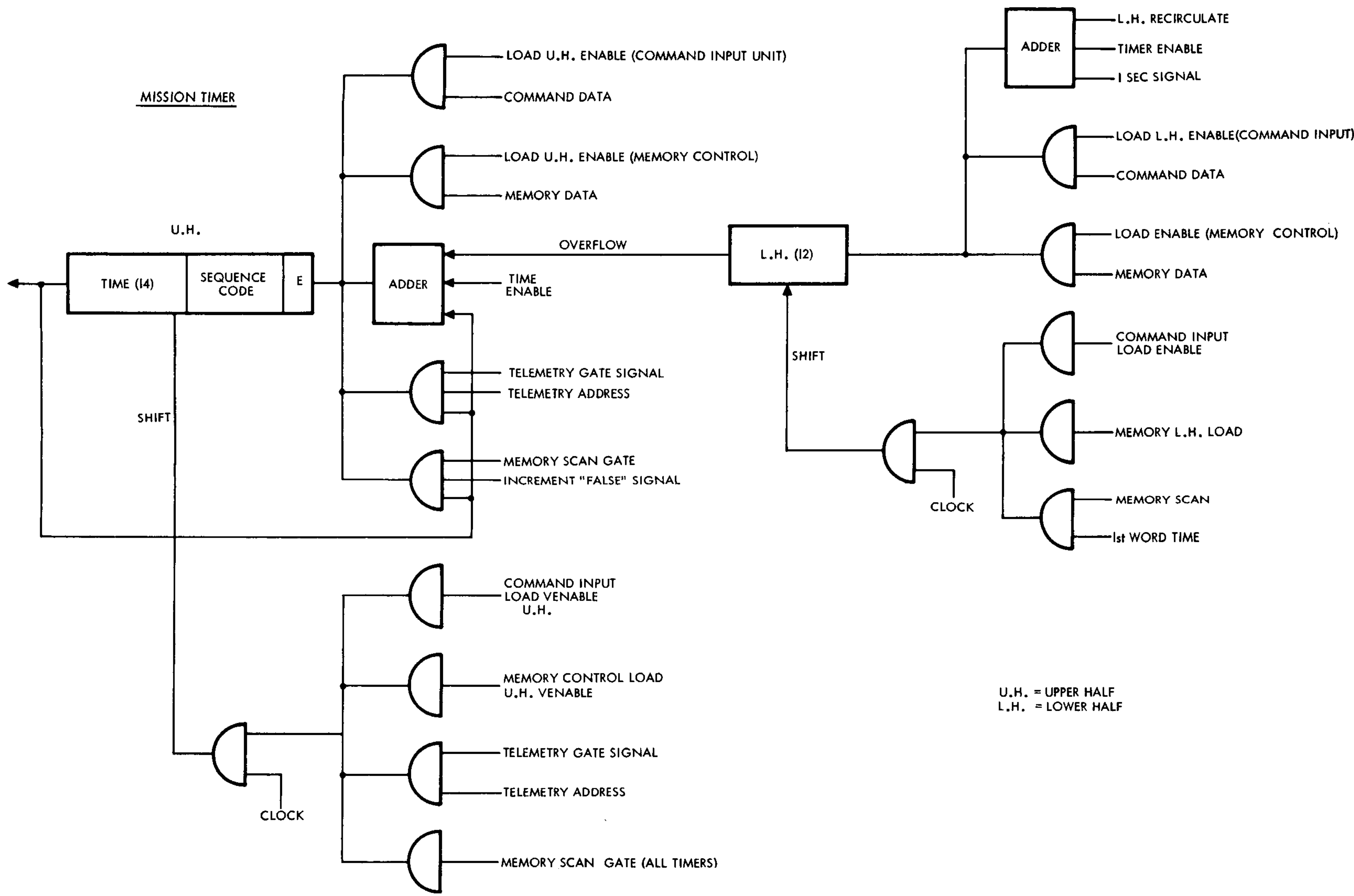


Figure 6-5
THE MISSION TIMER has two sections together with selection logic controlling loading, recirculation, and incrementation.

The contents of the two sections of the mission timer are read out in sequence to the telemetry register during computer and sequencer verification. In addition, the mission timer may be read out to provide time tags for recorded science data.

During prelaunch verification, the command and sequencer may be operated in a speedup mode by the operational support equipment. In this mode, the upper half of the mission timer cycles in 4.5 hours by incrementing at 1-second intervals. Hence, major events over the entire mission, associated with the timer, may be verified in this period.

6.3.2 Maneuver, Orbit and Science Timers

The maneuver, orbit and science timers (Figure 6-6) are essentially identical except that the orbit timer is normally incremented at 0.25 Hz and the maneuver and science timers at 1 Hz. The three timers provide for a sequence code and a range of 2^{14} increments. The maneuver and science timers with a 1-second granularity provide a 4.5-hour cycle and the orbit timer with a 4-second granularity provides an 18-hour cycle. However, the maneuver timer for certain orbital sequences may also be driven by the 0.25 Hz signal.

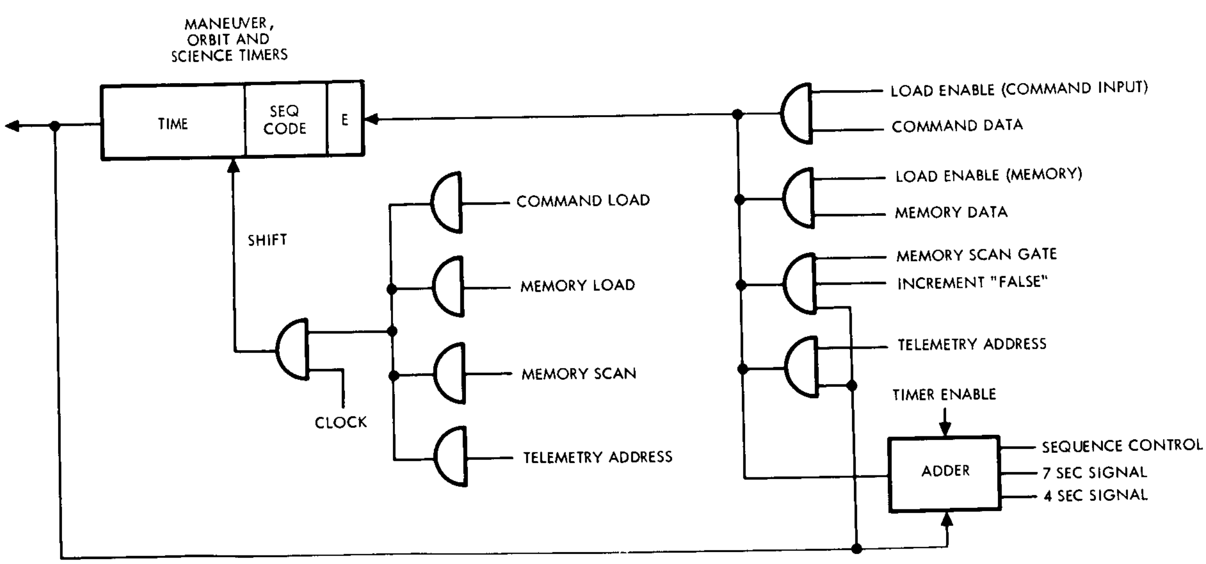


Figure 6-6

CONTROL LOGIC FOR THE MANEUVER, ORBIT, AND SCIENCE TIMERS is essentially identical for each of the three timers. All timers are implemented as simple shift registers and are incremented by circulation through a one bit adder.

Sequences controlled by these timers are identified by their associated sequence codes. These codes and an initial time setting (usually zero) are loaded into the respective timers from stored program commands. Alternately, the timers may be loaded from the ground via the command unit. In addition, special sequences are provided by an immediate sequence code and time preset in the orbit and science timers in response to a solar flare signal.

Each timer is controlled by an enable bit which may be set immediately, by some external event or by ground command. The timers are inhibited from counting until their associated enable bits are set.

The orbit and maneuver timers are physically associated with the sequencer memory and are read simultaneously with the main memory. Incrementing occurs during the first word time of a memory scan period. The mission timer and the science timer use integrated circuits and are logically synchronized with the memory readout. As with the mission timer, the maneuver, orbit and science timers are read in sequence into the telemetry register during computer and sequencer memory dump.

6.3.3 Command Input

The command input unit (Figure 6-7) accepts serial messages from the command subsystem, which are in turn received via the RF or hard line links. The command input section performs a parity check on the message and routes it to the indicated destination: the computer and sequencer memory; mission, maneuver, orbit or science timers; function generator, and accelerometer counters. Logic in the command input unit directs the 20 bits of data to the section indicated by the class and address bits. In addition, special identification codes enable unique functions such as the enable/disable of stored program commands or timers. The input message formats and special functions are shown in Figure 6-8.

The command input unit contains a 32 bit register for temporary storage of the incoming word. Data is received from the command system at an 8 bits per second rate and it enters synchronized by pulses supplied by the command system. Upon completion of transfer and message validation by the command system, an enable pulse is received and parity is verified by the command input unit. During the next input gate time in

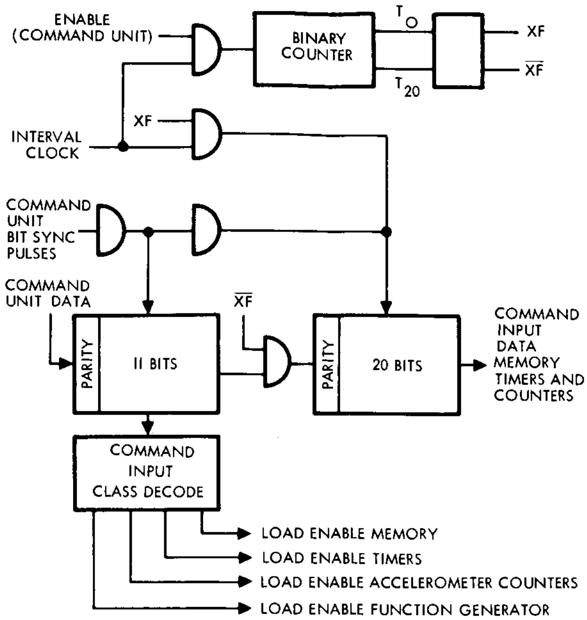
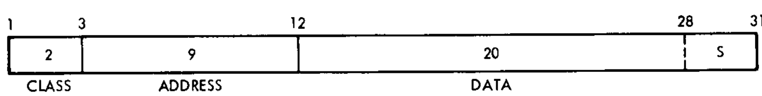


Figure 6-7

THE COMMAND INPUT ASSEMBLY: receives and processes ground commands for the primary sequencer. An identical assembly performs the same function in the backup sequencer. Ground commands are initially process by the command subsystem before they are fed to the computer and sequencer subsystem command input assemblies.



MESSAGE CLASS	CLASS FIELD	ADDRESS FIELD	DATA FIELD	SPECIAL FIELD	DATA DESTINATION
	(1,2)	(3 - 11)	(12 - 31)	(28 - 31)	
	00	-	-	-	NO ACTION
1	01	$0\text{-}256_{(10)}$	-	X	MEMORY STORAGE
2	10	1	-	X	MISSION TIMER L.H.
		2		X	MISSION TIMER U.H.
		3		X	MANEUVER TIMER
		4		X	ORBIT TIMER
		5		X	SCIENCE TIMER
		6		X	ACCELEROMETER CONTROL NO.1
		7		X	ACCELEROMETER CONTROL NO.2
		8		X	FUNCTION GENERATOR
3	11	X	X	0	ENABLE ALL MEMORY WORDS
		X	X	7	DISABLE ALL MEMORY WORDS
		$0\text{-}256_{(10)}$	X	1	ENABLE SELECTED MEMORY WORDS
		$0\text{-}256_{(10)}$	X	6	DISABLE SELECTED MEMORY WORDS
		1-8	X	2	ENABLE SELECTED CLASS 2
		1-8	X	5	DISABLE SELECTED CLASS 2

Figure 6-8

THE COMPUTER AND SEQUENCER SUBSYSTEM INPUT COMMAND WORD FORMATS are shown above. Message class 1 provides data for storage in selected memory locations. Class 2 provides ground data loading of the indicated assemblies. Class 3 messages provide the enable/disable functions specified by the least significant bits of the data field called the special field.

the computer and sequencer cycle, the input word is transferred to its ultimate destination. If the input word is a special function, the operation is performed. When a full message is received an enable signal from the command input unit must occur for final processing of the incoming command.

When the input has been validated, final processing for each class of message during the input gate time occurs as follows:

Class 1 The address portion of the input register is fed to the memory address gates and a one-word memory cycle is initiated. The data portion of the input register is shifted at the memory bit rate and data is stored in the indicated address.

Class 2 The address portion of the input register is decoded, indicating one of the seven computer and sequencer registers. The indicated register receives the data at the memory bit rate.

Class 3 These messages are special functions and are indicated by the three least significant bits in the data portion of the input word. If the command indicates the enable or disable of all memory words function, the entire memory is cycled with the enable bit in each word set to 1 or 0, depending on which function is present. For enable or disable of selected memory words, the address of the memory word is supplied and the enable bit of that word is either set or reset. The memory, in this case, is cycled for one word time only. For selected Class 2 functions, the registers indicated by the code in the address portion of the input word (as in Class 2 inputs) are enable or disabled.

6.3.4 Memory System

The sequencer memory (Figure 6-9) is a nonvolatile, ferrite core unit that operates in the coincident current mode. It is organized for bit serial access and contains 512 twenty-bit words. Memory cell address originates in a 9-bit address counter during memory scan, in the command input unit during input message store, and in the telemetry word address counter during computer and sequencer verification.

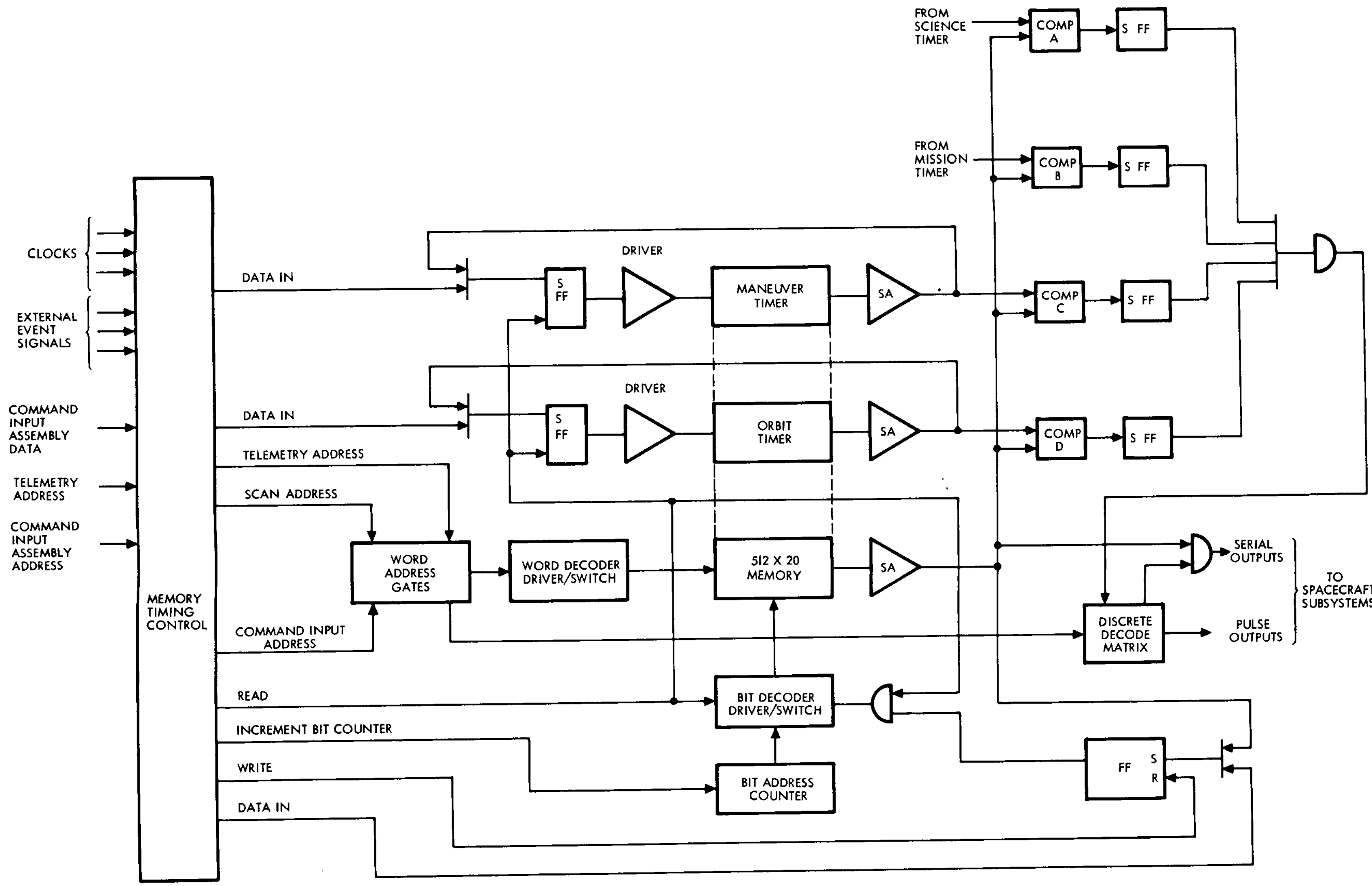


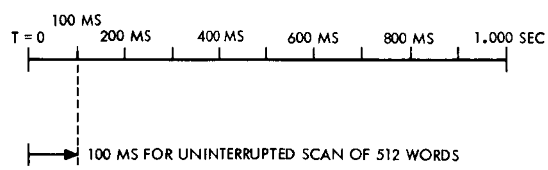
Figure 6-9

THE MEMORY AND CONTROL ASSEMBLY makes use of four one-bit comparators that receive inputs from the seperate mission and science timers and the integral maneuver and orbit timers. The maneuver and orbit timers are implemented as single word core registers associated with the memory bit address selection. As bits of each word are read from the central memory, the corresponding bits of the timers are also read. The comparators test for full word coincidence required for a stored program command output.

The maneuver and orbit timers are each contained as one-word core registers that are selected by the bit address only and read simultaneously with every access of the main memory.

The memory timing and control is supplied by a counter that divides the master 819.2 kb/sec clock by 8, generating a time period defining a memory cycle or one bit time. Additional control logic generates the memory cycle initiate, memory enable, and read/write control signals. The memory timing counter cycles in 9.77 microsec (102.4 kHz). A decoded signal at this rate is used for an internal sequencer clock.

The memory scan period, initiated by the 1 pps signal, is defined by one complete cycle of the word address counter, which in turn, requires a complete bit counter cycle for each increment. As each memory bit is read, the corresponding bits of the maneuver and orbit timers are read. In addition, the contents of the upper half mission timer and the science timer are cycled. Each memory bit is compared simultaneously with the corresponding bits of the four timers. If a full word coincidence occurs (including the enable bit), the bit and word counters are temporarily inhibited and the word address is decoded for a command output. The timing of the sequencer operation over a 1 second period is shown in Figure 6-10.



- ADD 50 MS FOR EACH DISCRETE COMMAND ISSUED
- TIMERS INCREMENTED DURING FIRST WORD TIME
- BIT TIME = 9.77 MICROSEC
- WORD TIME = 195.3 MICROSEC
- PORTION OF 1-SEC PERIOD REMAINING AFTER SCAN. AVAILABLE FOR COMMAND INPUT AND/OR TELEMETRY OUTPUT.

Figure 6-10

TIMING OF THE SEQUENCER OPERATION over a one second period is shown above. 100 ms are required to scan the 512 words of memory. 50 ms are added for each coincidence that is found during a scan. The remaining portion of the period is allotted for command input and/or memory dump when the latter is commanded.

6.3.5 Command Output

When a full word coincidence occurs, the memory address lines are gated to the command decode matrix (Figure 6-11) where the 9-bit binary address code produces a unique output of the 512 available. A 50 msec pulse is generated, which activates the command output driver affected. At the end of this period, memory scan is resumed.

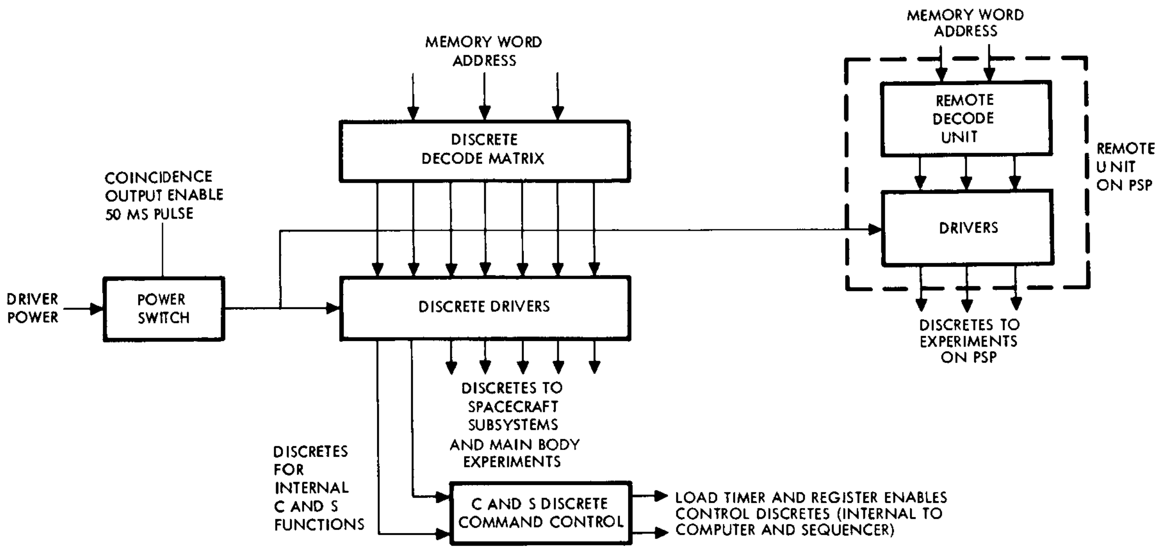


Figure 6-11

COMPUTER AND SEQUENCER SUBSYSTEM COMMAND OUTPUT ASSEMBLY generates the stored program commands and operates independently of the decoder in the command subsystem. A portion of this assembly is implemented remotely on the PSP but receives inputs and generates outputs as though it were an integral part of the assembly.

Certain commands signify that a serial message is to follow. These messages are stored in memory locations immediately adjacent to the associated command. For messages internal to the computer and sequencer (i.e., data for the timers, counters or function generator) the 50 msec command output period is not required and the command is issued just long enough to set a message control flip-flop. With the flip-flop set, the memory scan resumes and the next word read from memory is directed to the section indicated by the particular command flip-flop. If the command signifies the message is to be fed to a unit external to the computer and sequencer, the command is produced as usual with the message following.

The mission and orbit timers have a granularity greater than the memory scan cycle, (i.e., the memory is scanned more than once during each interval of the timer). To avoid repeated command outputs during such intervals, inhibit logic prevents coincidence occurring more than once during each timer interval.

A section of the command decode matrix that produces commands for experiments located on the planetary scan platform is remotely located on the platform. This remote decoder receives and decodes the memory address code in the same manner as the portion within the computer and sequencer. This technique affords a flexible interface to the science instruments on the platform and also standardizes and limits the size of the cable to the platform.

6.3.6 Accelerometer Counters

Two redundant accelerometer counters (Figure 6-12) are provided to time-out engine burn periods. The operating counter is selected by

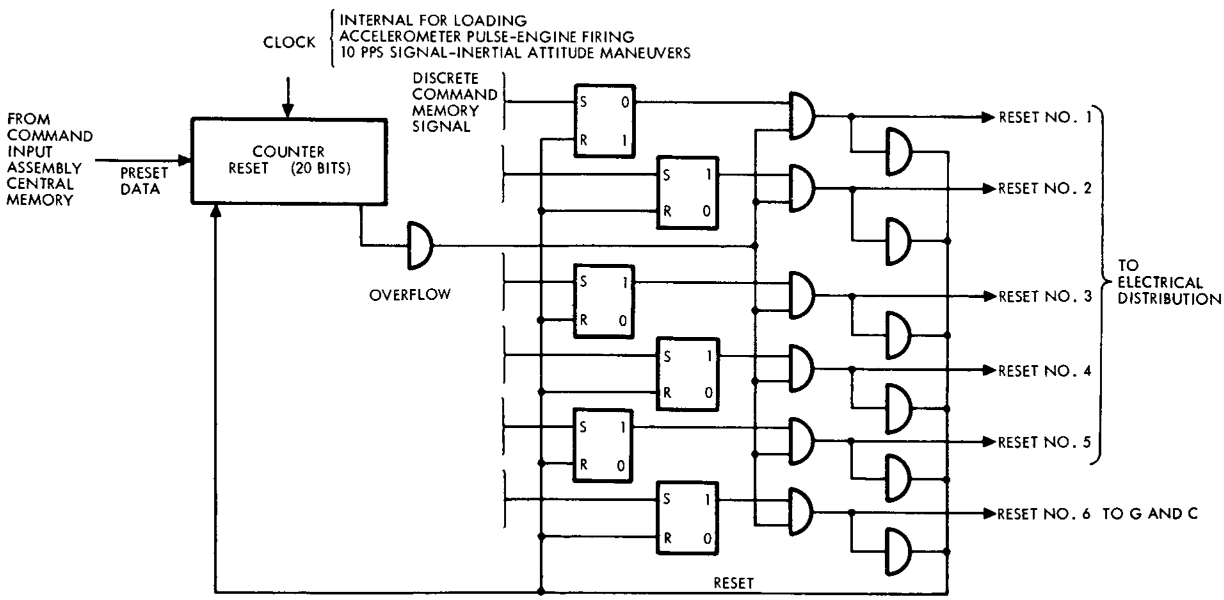


Figure 6-12

THE ACCELEROMETER COUNTER consists of two redundant systems implemented in the primary sequencer. The counters may be loaded and operated from the ground independently of the central sequencer memory. The reset outputs 1 through 5 are fed to the electrical distribution subsystem where they initiate engine shutdown following the accumulation of the proper number of accelerometer pulse outputs. Reset 6 is fed to the guidance and control subsystem initiating an attitude maneuver termination following accumulation of the proper number of 10 pps signals generated in the clock divide assembly.

commands issued from the computer and sequencer memory. The selected counter is preset with the complement of the pulse count representing the velocity increment to be obtained. Pulses are received from the accelerometers and are accumulated in the counter until an overflow occurs at which time a reset pulse is generated initiating engine shut-down. As a backup to the accelerometer counter, the computer and sequencer can issue a command for engine shut-down by preprogramming a time for maximum engine operation as part of the maneuver sequence.

In a similar manner, the accelerometer counters provide a fine-grained time period for inertial attitude maneuvers. The selected counter is loaded with the complement of the time period desired and stepped at a fixed 10 Hz rate furnished by the clock-divide section. The overflow generates the maneuver termination signal.

The velocity correction requirement for the vehicle ranges from 1 to 1905 meter/sec with a defined maximum acceleration of 3.2 g. The quantization is 0.46 cm/sec pulse. Hence, the pulse count can range from 219 to 417,000. The maximum pulse rate is 4945 pulses/sec. For the given granularity and maximum velocity increment required, a 19-bit counter is adequate without changing the scale factor in the accelerometer. Since a 20-bit data word is permitted in memory, this size is specified for the velocity increment accumulator register.

For inertial attitude maneuvers, pitch, yaw, and roll rates are fixed at 0.2 deg/sec. To meet maneuver accuracy requirements, a time base accurate to 0.1 sec must be supplied for a maximum maneuver of 180 degrees. This results in a maximum count at 10 Hz of 9000 or approximately 2^{14}, well within the range of the counters.

Two commands are necessary to activate the counter. One command is required to load the counter initially with a predetermined number. Another is needed to set an enable flip-flop which selects one of five reset gates for velocity correction maneuvers or the single reset gate for the inertial attitude maneuver. Following the counter overflow, the reset pulse is issued via an enabled gate. This pulse also resets the enable flip-flop and the counter.

The counters may be preset by stored program commands or from the ground. Two separate counters and gate systems are provided and each is separately addressable from memory and the ground.

6.3.7 Function Generator

The function generator (Figure 6-13) uses approximating linear functions to generate pointing commands for the high gain antenna on a continuous basis through the entire mission. The unit accepts initial angular data either from stored program commands or from the ground and issues command data to the hinge angle gimbal register of the antenna.

The pointing angle function, given in spacecraft body coordinates, is approximated by n linear segments, θ (t), where n< 10.

Let $\theta(t_I) = \theta_I$ be the initial angle for the ith segment and let $\Delta\theta_I$ be the fixed angle change in that segment for the fixed increment time Δt. Assume the initial time for each segment is given in terms T_i of the mission clock.

At the initiate time T_1 for the first linear segment, shift θ_i and $\Delta\theta$ into the function generator, and using the computation "start" command, issue the angle θ to the high gain antenna cone gimbal register.

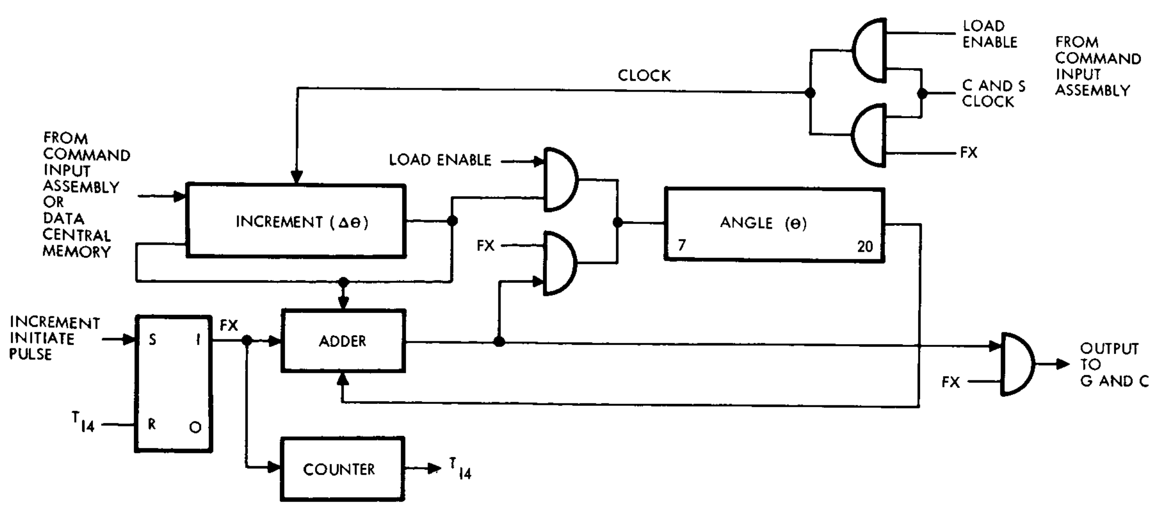


Figure 6-13

THE FUNCTION GENERATOR periodically the contents of the two sections are circulated and summed. The new sum is loaded into the angle register. At the same time the new angle data is fed to the antenna control in the communication subsystem. The sections are loaded with initial data from either the central memory or from the ground via the command input assembly.

At $T_1 + \Delta t$ form $\theta_1 + \Delta\theta_1$, store this quantity in the θ register and issue the quantity to the gimbal register. Repeat for $\theta_i + 2\ \Delta\theta$, etc. At T_2, shift θ_2 and $\Delta\theta_2$ into the registers and continue the process.

The length for the time tag, T_1, is 14 bits. The time is referenced to the high order bits of the mission timer. The time increment, Δt, is obtained from the low order bit of the high order 14 bits of the mission timer. This is sampled by the function generator on a continuous basis to establish times to issue gimbal angle data. The bit length for θ_i is 14 and the bit length for $\Delta\theta$ is 6.

Figure 6-13 shows a block diagram of the function generator. Two main registers, "angle" and "segment", contain the present pointing angle, θ, and the current $\Delta\theta$ increment. At each computation time obtained from the mission timer, $\Delta\theta$ is added to θ, a control signal is sent to the antenna, and the new data of θ is issued as a serial message. This process continues throughout the mission and occurs on a regular basis. Following a control signal from the memory (or from ground), new data is loaded into the two registers and additions continue. An inhibit signal generated by specific sequence codes prevents any outputs to the antenna during maneuver conditions.

6.3.8 Telemetry Data Register

The telemetry register (Figure 6-14) of the computer and sequencer provides buffering for data during memory content verification. Words are loaded into the register from the memory, timers, function generator and accelerometer counters in a sequential fashion. Addressing is provided by a 9-bit memory address counter and an 8-state counter for sequencing the remaining timers and registers. The memory address counter feeds the memory word address gates.

Over half of each 1-second period is allocated as "telemetry gating" time. (See Figure 6-10.) Upon ground command or stored program command initiation of memory dump, the telemetry subsystem sends a signal to the computer and sequencer subsystem that indicates telemetry is ready.

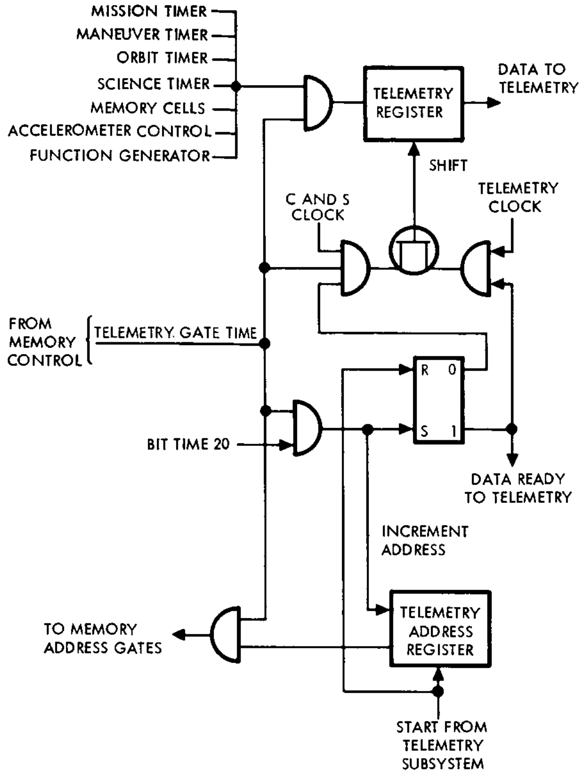


Figure 6-14

THE TELEMETRY DATA REGISTER includes a telemetry address register that is reset to zero at the beginning of a memory dump operation and is incremented as each word is read out of the memory and sampled by telemetry. Addresses are also provided for the timers and special registers indicated. Telemetry sampling occurs at the prevailing transmission bit rate.

This signal resets the telemetry address register and initiates the memory dump process. When the next telemetry-gate time occurs, the first word in memory is read into the telemetry register and the computer and sequencer ready signal is sent to the telemetry subsystem. The telemetry subsystem responds by sending clock pulses at the current bit rate to shift the word out of the computer and sequencer. The address counter is incremented and the next word is read from memory. This process continues until the memory address counter completes a full cycle and the timer/register address counter is cycled.

If, due to a low current bit rate, the process is not completed during a single telemetry-gate time, it is interrupted while memory scan occurs and then resumed from where it was interrupted.

The telemetry and data storage subsystem samples the data 6 bits at a time. Consequently, the first bit of every 6 bit sample contains a "1" to differentiate between actual zero data and no-sample words. The "1" bit is inserted between every 5 bits as they are loaded into the telemetry register.

6.3.9 Command Events Counter

The command events counter (Figure 6-15) provides a tally of the stored program command issued by the computer and sequencer. A comparison can then be made between the number of commands actually issued and the number expected.

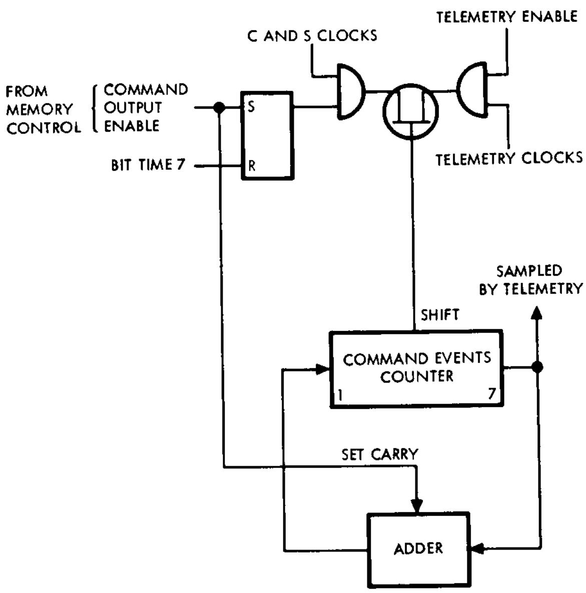


Figure 6-15

THE COMMAND EVENTS COUNTER provides a limited check on the Computer and Sequencer operation by tallying the number of stored program commands issued. The contents of the counter are sampled periodically as engineering data. The counter is reset to zero following each sample.

The command events counter is pulsed by the stored program command enable signal that is generated when a memory word/timer coincidence occurs. The counter is capable of accumulating up to 128 counts between samplings by telemetry. Sampling occurs at regular intervals as part of the engineering telemetry. The register is sampled in parallel and is reset to zero immediately following each occurrence.

7. ELECTRICAL POWER

7.1 SUMMARY

The electric power subsystem converts solar energy into electric power for use by the planetary vehicle subsystems. Nickel-cadmium batteries store electrical energy when a surplus is available from the array. During periods when the array is not receiving sunlight, the batteries provide power to the subsystems.

The maximum power demand is 808 watts. This occurs in Martian orbit at 1.62 AU. At that time the 226-square-foot fixed solar array can deliver 836 watts. The array has no deploying or moving parts. The upgraded Voyager spacecraft may also not need deployable solar array panels.

Nickel-cadmium batteries were chosen because their life expectancy is the longest of competitive types. In fact, these are currently the only batteries capable of performing upgraded missions of 30 months.

Regulation of solar array and battery power within the power system is carried out by a power control unit uniquely suited to the Voyager mission profile. It boosts the voltage in the early part of the mission and reduces it if required during the later part of the mission such that the solar array operates at nearly the maximum power point in all mission phases. This regulator increases in efficiency as the mission proceeds. It also makes less demands on the solar array and battery than alternative systems we have examined. Furthermore, it is lighter than the alternative regulator systems.

The power subsystem delivers unregulated direct current at between 37 and 50 volts to the spacecraft subsystems. Our tradeoff studies (see Appendix S) showed that weight and efficiency differences between different AC and DC power distribution systems were very small. However, past experience and requirements for electromagnetic compatibility strongly favored direct current distribution.

Each spacecraft subsystem will be equipped with its own DC-to-DC converter, physically located within the subsystem. However, these converters will be provided as part of the power subsystem in response to the detailed specifications of each user subsystem.

Load control signals for the removal of nonpriority loads at critical periods will be provided by the power subsystem. The design and implementation of this function will be carried out when detailed specifications from user subsystems are available.

The preliminary specification for the subsystem appears in Figure 7-1, and its general arrangement on the spacecraft is shown in Figure 7-2.

7.1.1 Constraints

The Voyager mission described in the JPL specification extends for 14 months after launch, including six months in Mars orbit.

After liftoff the maximum period before sun acquisition will be 150 minutes. Subsequent maneuvers that require loss of solar alignment,

PRELIMINARY SPECIFICATION

Electrical Power Subsystem

Performance Characteristics

Main Bus Voltage	37 to 50 volts DC
Solar Array Power	1.67 au 787 w 1.62 au 836 w 1.00 au 1426 w
Battery Energy	1968 watt-hours
400 Hz Two-Phase for Antenna and Planetary Scan Platform Drives	50 v RMS ± 2%
Synchronization signal (Clock)	819.2 kHz

Physical Characteristics

Subsystem weight	443 pounds
Solar Array Area	226 square feet

Components

44,640 N-ON-P Solar Cells
Three 16 Ampere-Hour Nickel-Cadmium Batteries
Power Control Unit
400 Hz Inverter
DC-DC Converters

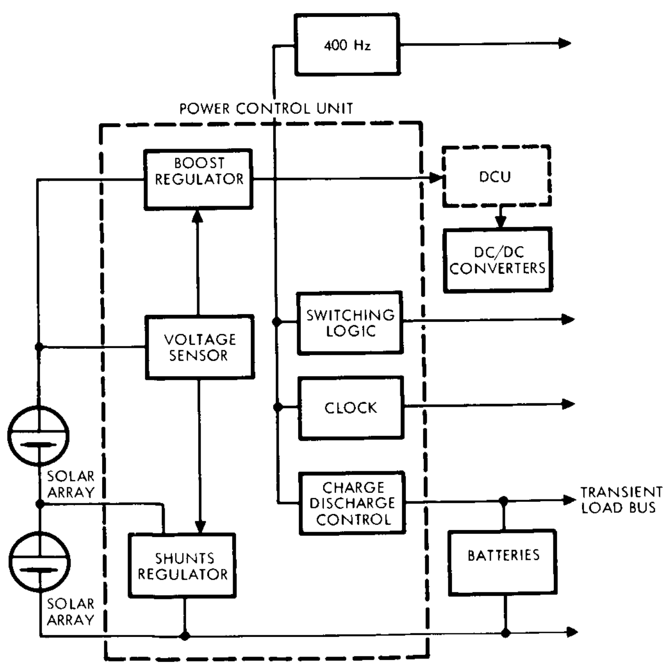


Figure 7-1

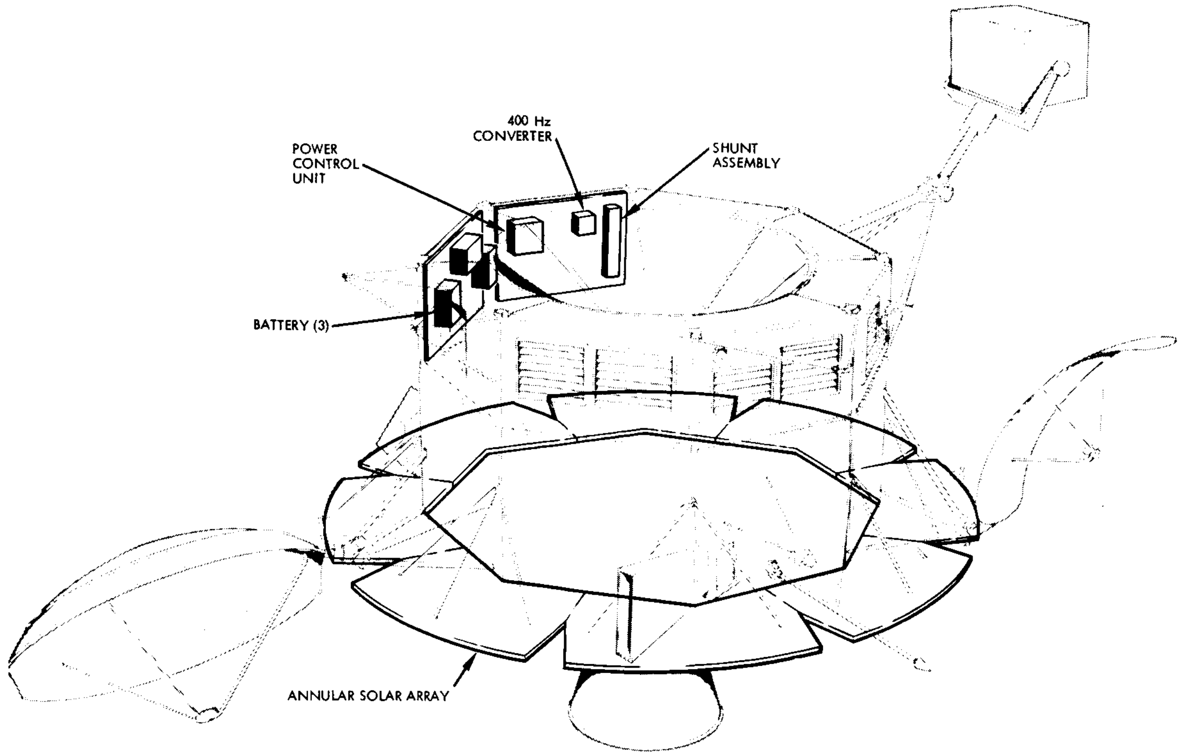


Figure 7-2
The electrical power subsystem on the recommended spacecraft derives its primary supply from the annular array and from additional solar panels (not shown) on the equipment module. During maneuvers or solar occultation, power is supplied by three nickel-cadmium batteries.

such as trajectory correction, orbit insertion, or orbit trim, will mean loss of solar power for not more than 120 minutes. From Mars encounter till the end of six months orbital operation, the spacecraft will be from 1.62 to 1.67 AU from the sun.

Eclipses do not start until at least one month after Mars orbit insertion and do not last longer than 2.3 hours, in the worst case.

The electric power requirements of the planetary vehicle subsystems during the various mission phases are shown in Table 7-1.

7.2 FUNCTIONAL DESCRIPTION AND PERFORMANCE

The main components of the electric power subsystem are the solar array, three nickel-cadmium batteries connected in parallel, the power control unit, a 400-Hz inverter, and eight DC-to-DC converters, one for each user subsystem. A functional description of each component is presented below.

Table 7-1. Voyager Average Power Profile (Watts) Summary

Subsystem	Configuration/ Usage	Form	Prelaunch	Cruise (1)	Trajectory Correction	Cruise (2)	Orbit Insertion	Orbital Operation PV	Orbit Trim (1)	Capsule Separation	Orbit Trim (2)	Orbital Operation Sun Oriented	Orbital Operation Eclipse
S-Band Radio		50V 400 Hz	25	180	180 60	180	180 60	180	180 60	180	180 60	180	180
Guidance and Control		50V	51	60	205	60	205	65	205	60	205	65	60
Thermal Control		50V		62	72	62	72	52	42	52	52	52	52
Capsule Support		50V		200		200		200		50			
Science		50V 400 Hz	6	6	6	6	6	21 14	5	5	5	39 14	84
TM and Data Storage		50V	7	7	21	7	21	32	21	26	21	35	41
Command and Sequence		50V	41	41	41	41	41	41	41	41	41	41	41
Distribution		50V	7	7	7	7	7	7	7	7	7	7	7
Power (PCU) Inverter		50V 50V	10	10	10 18	10	10 18	10 3	10 18	10	10 18	10 3	10
Subtotal			147	573	620	573	620	625	589	431	599	446	475
Harness Loss			1	6	6	6	6	6	6	4	6	4	5
Total	5 TV System LEM and C-1 Engine		148	579	626	579	626	631	595	435	605	450	480
Total	Film/TV System LEM and C-1 Engine		148	579	626	579	626	658	595	435	605	477	524
Battery Charging			-	72	-	72	-	(5)110	-	-	-	210	-
Power Control Losses			-	45	-	45	-	-	-	-	-	-	-
5 Percent Margin			-	35	-	35	-	40	-	-	-	35	-
Total array or battery power required			148	731	626	731	626	808	595	435	605	722	524

*The Mars mission average power profile indicates that the critical requirement for the solar array occurs during the orbital operations after insertion into Martian orbit (830 watts). This is the time when the capsule is still on, the planetary mapping has started, and the batteries have to be recharged after the previous maneuver. The critical requirement for the batteries is during a solar eclipse season in Martian orbit. Depending on the type of orbit chosen, it will either require a 61 percent discharge for a maximum of 120 cycles or less than 50 percent for up to 300 cycles. The power control losses shown in the tabulation concern only the boost regulator. The propulsion subsystem represents only a transient load since the maneuvers are very short compared to the total mission. The recharge of the battery at 90°F after insertion into Martian orbit may take up to 23 hours.

The power subsystem has two main modes of operation during the entire Voyager mission (see Figure 7-3). The first extends up to the Mars orbit eclipse season; the second occurs during eclipse season. In the first mode the solar array provides most of the electrical energy for spacecraft subsystems. The battery supplies only a small amount of energy during short periods when demands exceed the solar array capacity. Although operation in this mode affects the subsystem design, the parameters become critical during the second mode (Mars orbit operation). This second mode is characterized by more equal sharing of the load between the solar array and the battery. Both the solar array and battery sizing criteria are provided during this phase. However, these two basic units of the power subsystem cannot be designed independently, and the recommended design described here takes into account the interactions of the solar array and battery throughout the mission.

The maximum power demand occurs at this time (see Table 7-1) and the array has to be sized to meet this. The demands of the planetary

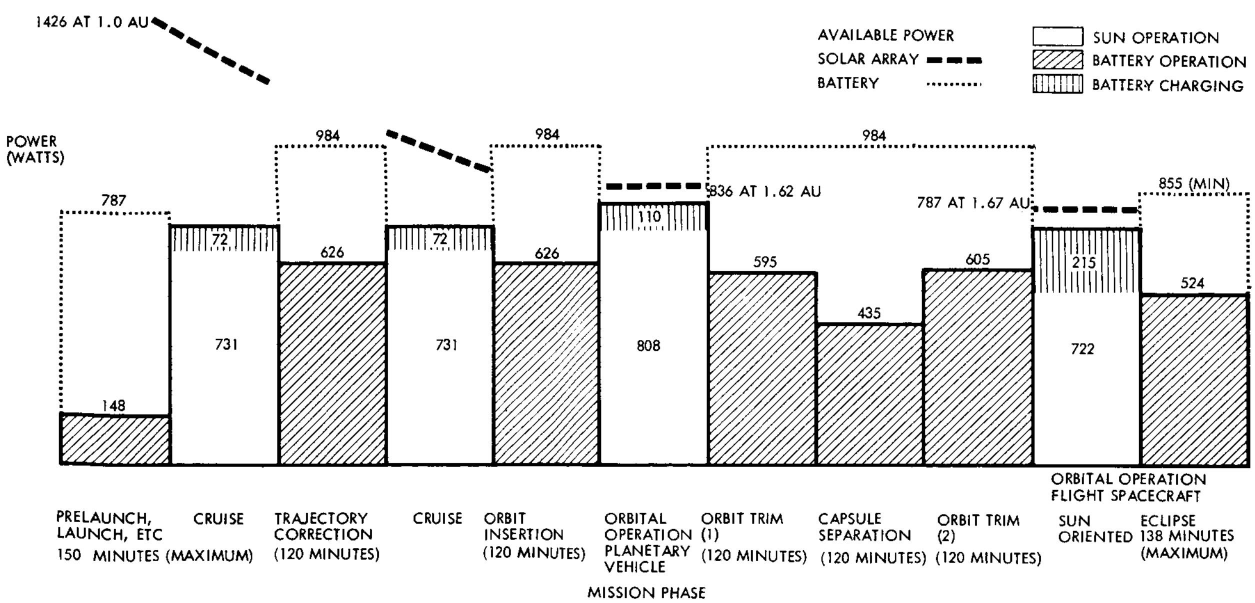


Figure 7-3

THESE REPRESENT THE MAXIMUM POWER REQUIREMENT for each phase. As shown, the phase dictating the size of the Solar Array is the planetary operation after insertion into the Martian orbit. Although the battery discharge appears large during this maneuver, because of the intended use of the nickel-cadmium battery, this can be tolerated and does not affect the battery size. Battery size is determined by the continuous cycling phase occurring during the eclipse season after the release of the landing capsule. Reconditioning of the battery, with a full recovery of its capacity, can be accomplished-prior to the start of the eclipse season-on command.

vehicle, including charging, reach a peak of 808 watts after orbit insertion (assuming that the planetary science payload is not turned on before the vehicle is checked out and its orbit determined). At this time the solar array of the recommended spacecraft is capable of providing 836 watts to the planetary vehicle subsystems.

7.2.1 Solar Array Control

Because the array is sized to meet the critical power requirement at the end point of the mission, there is power to spare in the early parts of the mission. However, the transit from earth to Mars is not only accompanied by a decrease in solar power; at the same time, the solar array temperature is dropping, causing the output voltage to rise (Figure 7-4). The rise is such that when the array voltage is optimized for Mars orbit requirements, the array voltage near earth is far too low to charge the spacecraft battery directly. A boost regulator raises the array output voltage to a level that can provide battery charging.

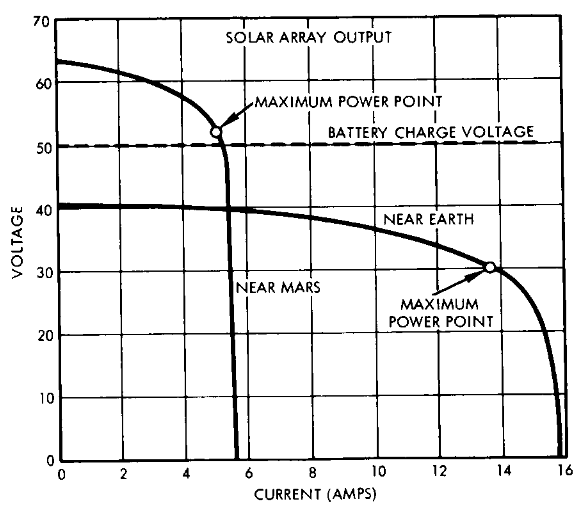


Figure 7-4

SOLAR ARRAY OUTPUT CHARACTERISTICS. The curves shown correspond to the two extremes of expected operation of the Solar Array as far as distance from the sun is concerned (1 AU for near earth operation and 1.67 AU for operation at the Mars orbit when the distance of Mars to the sun has reached a maximum). The PCU design described takes advantage of these characteristics in that although the net power output of the array decreases with time, the voltage increases, making the PCU more efficient and minimizing boost regulator losses. When the control has switched to the shunt assembly the power losses are nearly zero when the load has reached a maximum. (This figure shows performance of the insulated array only).

Then as the spacecraft recedes far enough away from earth for the solar array voltage to be sufficient to charge the batteries directly, the boost regulator can be taken out of the circuit. Its losses are thereby removed. To limit the voltage subsequently supplied by the solar array to the battery and the other spacecraft subsystems, a shunt regulator is brought into the array circuit. This regulator becomes operational at the point where it can be most effective, for its losses are at a minimum when the load approaches the solar array power output.

In the shunt regulator, excess capacity is dissipated by means of shunt elements that cause array sections to operate at off-optimum points. The shunt regulator includes one linear element and several switching elements. Initially, the linear shunt element, which is essentially a linear power amplifier, shunts current from the lower section of the tapped array up to a maximum of about 5 amps. As further array capacity must be dissipated, the switching shunt elements become active. These elements can either short out an array section completely or allow an array section to deliver its full power capability. The "on" points are set so that the first switching element shorts when the total shunt current requirement is 3 amps; the second shunt element shorts when the total shunt current requirement is 5 amps, continuing on until the N switching elements are saturated, at which time the linear element can then shunt an additional 5 amps. During the sequential turn on cycle of the switching shunt elements, the linear shunt element shunts currents of up to 3 amps.

A block diagram of the full power subsystem showing the combined regulator scheme appears in Figure 7-5.

Tradeoff analysis (see Appendix R) shows that the boost-shunt method of power regulation puts the lowest demands on the solar array and the battery and is also the lightest system.

In fact, this combined regulation scheme, using boost or shunt, gets progressively more efficient as the mission proceeds and the surplus power margin decreases. The boost regulator becomes more efficient as its input voltage approaches the desired output voltage, and the shunt regulator becomes more efficient as the load approaches the maximum solar array output power.

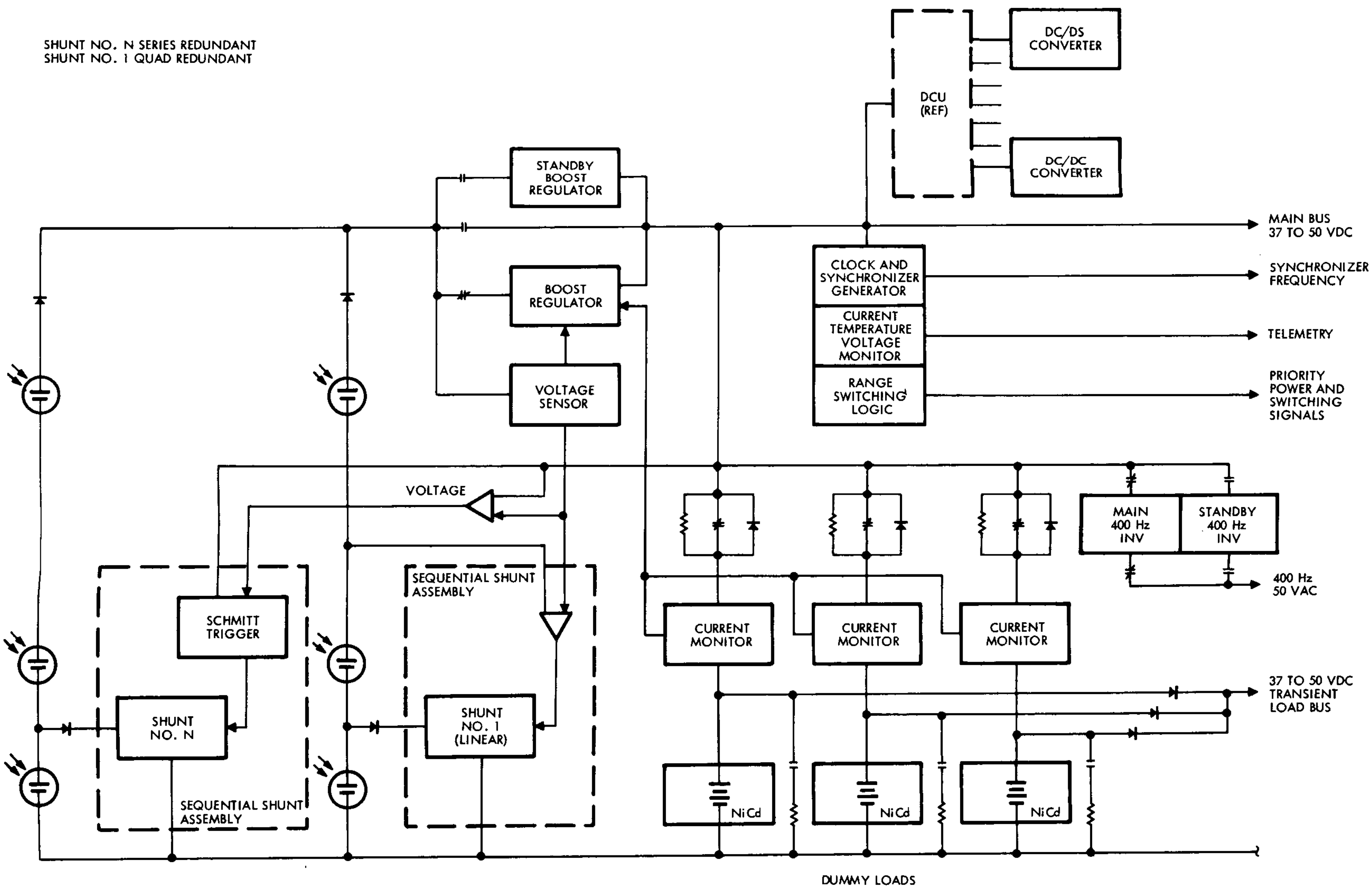


Figure 7-5

POWER SUBSYSTEM BLOCK DIAGRAM displays main features of system: Solar array output is regulated either by the boost regulator or the shunt assembly -- the former in the early stages when the voltage is too low to charge the batteries, and the latter when the solar arrays are at higher voltage. Also note the very simple battery charging circuitry, and the dummy loads used for battery reconditioning which can only be accomplished on command. Raw power is fed through the DCU to DC/DC converters which physically form part of the particular subsystem they supply power to. Switches shown indicate operation during early phases while batteries are charging at high rate.

7.2.2 Battery Control

Nickel-cadmium batteries have been chosen to supply power to all loads during solar eclipses and spacecraft maneuvers when the solar array is not facing the sun. Tradeoff studies resulting in the selection of nickel-cadmium are given in Appendix Q. Analysis of the power load mission profile (Table 7-1 and Figure 7-3) indicates that the dominant battery energy requirements occur during orbital operation at Mars (Table 7-2), specifically during solar eclipses which determine the battery design because of their magnitude and repeated cyclic occurrence.

During the eclipse season, the battery is being cyclically charged during sunlight operation and discharged during eclipse. In the worst case, the eclipse lasts for 2.3 hours, but these eclipses do not last for more than 120 cycles, and 61 percent discharge is permitted. Orbits with shorter eclipses also have more of them, such that the number of charge/discharge cycles for the battery can go up to 300. In this case the depth of discharge will never exceed 50 percent. To meet the spacecraft loads presented during this period (524 watts), the battery must deliver a stored energy of up to 1205 watt-hours. This figure, together with the 61-percent depth of discharge selected to meet the worst-case requirement of 120 charge-discharge cycles during the 14-month Voyager mission leads to a battery with an energy capacity of 48 amp hours (1968 watt-hour) weighing 150 pounds (see also Appendix Q, paragraph 1.6).

Table 7-2. Battery Energy Requirements

Mission Phase	Battery Output Energy Required (watt-hr)
Launch to Acquisition	370
Trajectory Correction	1252
Orbit Insertion	1252
Planetary Vehicle Orbit Trim	1190
Capsule Separation	870
Flight Spacecraft Orbit Trim	1210
Eclipses in Mars Orbit (maximum)	1205

Figure 7-6 shows the method of charging, discharging, and control of three nickel-cadmium batteries. When the spacecraft enters an eclipse with the batteries fully charged, a voltage monitor signal at about 47 VDC causes all batteries to be switched directly to the main bus. The batteries then supply power to the loads directly through quad-redundant relay contacts which minimize the power losses during discharge.

This condition continues until the eclipse period ends, when solar array power increases. At this time, the bus voltage is clamped to the final discharge voltage of the batteries. When the array power exceeds load requirements, excess current is supplied directly through relay contacts to the batteries. The batteries charge toward an upper voltage limit (of 47 VDC) compensated by temperature. Since the charge rates of the batteries will not be identical, a circuit limits the current to any single battery. As each battery reaches its limit, it is switched to trickle charge rate. The batteries remain in trickle charge until another eclipse occurs. While in trickle charge, the batteries can support any transient overload which exceeds the illuminated array capacity.

Pulse discharge of the batteries is accomplished with bypass diodes. A separate bus directly connected to the batteries is provided to supply 37 to 50 volts for transient loads, mainly from ordinance devices and

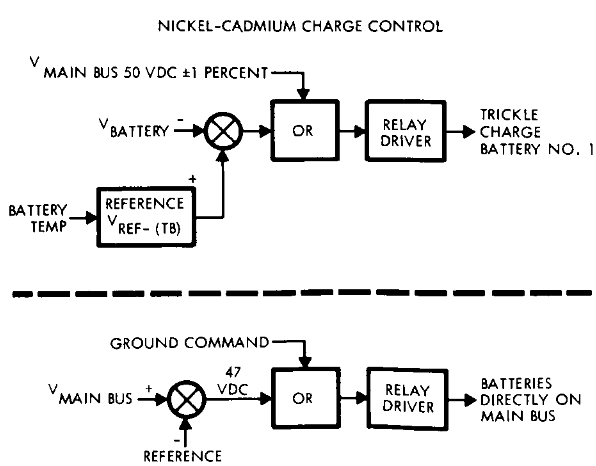


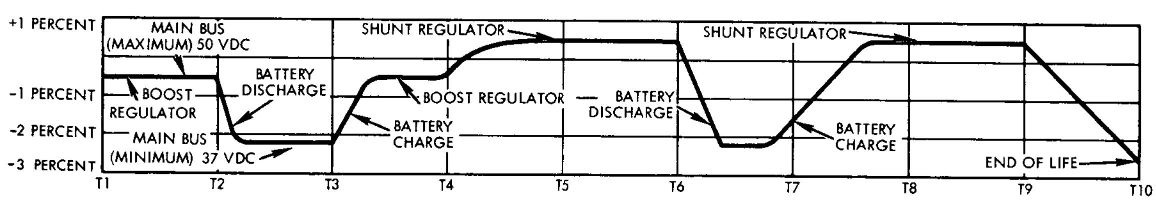
Figure 7-6
BATTERY CHARGE CONTROL provides either maximum current charge, trickle charge or discharge onto the main bus, depending on battery voltage, main bus voltage and Solar Array outputs.

gimbal motors. Figure 7-7 shows the main bus operating range and typical charge-discharge cycles.

A further feature included in the power subsystem is the capability to recondition the batteries prior to the major power drains that occur during Mars orbital operations. Nickel-cadmium batteries have a characteristic of losing stored energy when subjected to small repeated levels of discharge which is known as memory effect. This effect can be minimized by deeply discharging the battery and recharging it. The electric power subsystem contains the equipment necessary to accomplish this reconditioning upon receipt of the appropriate ground commands.

7.2.3 Load Control

A load control system within the power control unit will provide a signal to the DCU to disconnect nonessential loads when the current demanded from the power subsystem exceeds a preset limit. Maximum time to disconnect will be 10 milliseconds. Nonessential loads are disconnected when battery voltage decreases below a preset minimum for a time exceeding about 100 milliseconds. A block diagram of the load control system is shown in Figure 7-8.



TIME EXPLANATION

T1 - BATTERY IS FULLY CHARGED TO TEMPERATURE VOLTAGE LIMIT (50 VDC ±1 PERCENT) AND ARRAY IS SUPPLYING THE LOAD, BOOST REGULATOR OPERATING.

T2 - ECLIPSE HAS STARTED, I_{sa}, I_{LOAD} BATTERIES CONNECTED DIRECTLY TO THE MAIN BUS, I_{sa} DECREASES TO ZERO, BATTERIES SUPPLY THE LOAD.

T3 - START OF EMERGENCE FROM ECLIPSE, BATTERIES CHARGE.

T3, T4 - BOOST REGULATOR OPERATING.

T4 - BOOST REGULATOR NO LONGER OPERATIVE.

T5 - SHUNT REGULATOR OPERATING, EXCESS CURRENT SHUNTED.

T6 - ECLIPSE CYCLE.

T7 - END OF ECLIPSE, BATTERIES CHARGING.

T8 - BATTERIES CHARGED, SHUNT REGULATOR OPERATING, EXCESS CURRENT SHUNTED.

T9 - SOLAR ARRAY DEGRADES.

T10 - END OF LIFE.

Figure 7-7

RESPONSES OF POWER SYSTEM vary as solar array and battery outputs change during mission.

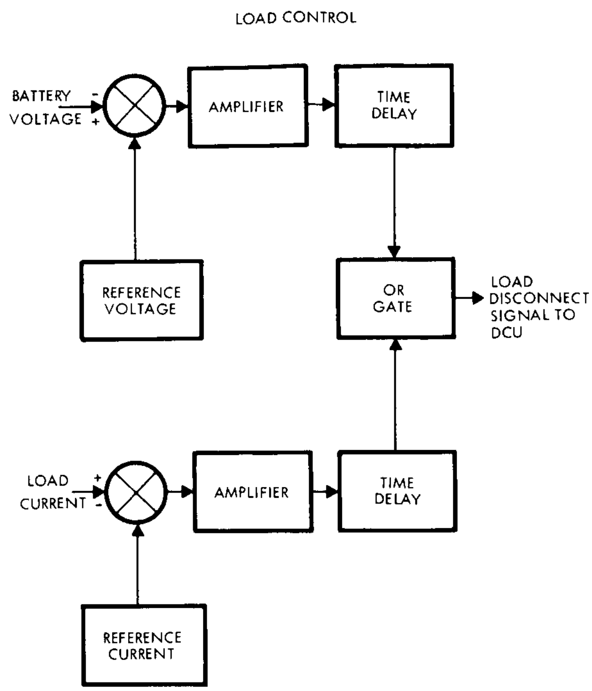


Figure 7-8
LOAD CONTROL UNIT sends signals to DCU to disconnect non-essential current or battery voltage levels exceed limits for more than 100 milliseconds.

7.2.4 Failure Modes and Redundancy

The power subsystem design includes many levels of redundancy to provide for continued operation in the event of a component failure.

7.2.4.1 Solar Array

The solar array is composed of silicon solar cells interconnected in a redundant series-parallel matrix. The basic building block is a submodule of three cells in parallel. These submodules are connected in series to form modules which are, in turn, connected in series to form a string. In the total array there are 132 such strings in parallel. Parallel redundancy of the strings is accomplished by this configuration and, in addition, the parallel redundancy of the submodules results in a total array which is essentially unaffected by single cell failures either in open circuit (most likely) mode or short circuit mode.

7.2.4.2 Batteries

The anticipated failure mode of a battery is either a cell open circuit or short circuit. If a cell goes open circuit, the whole battery is removed from further operation, reducing power system stored energy by 33 percent. If a cell shorts, the battery can continue to store energy, but delivers it back to the system at a reduced voltage. This reduces available stored energy, but by less than 33 percent since the remaining battery cells can still discharge to a reasonably low potential. The present design depth-of-discharge level with three batteries operating is 50 percent or less for 300 cycles or 61 percent for a maximum of 120 cycles. If a battery fails, this level of discharge can be increased with a slight degrading effect on cycle life of the battery. Alternatively, the total load power drawn from the batteries can be decreased during the deep discharge periods to retain discharge level near the 50-percent point. This change can be made by ground command.

7.2.4.3 Boos t Regulator

A failure of the boost regulator would result in inadequate power to the loads and inadequate battery recharging. This would probably be a mission failure and for this reason a second standby redundant boost regulator is switched in. This switching is implemented automatically but can also be commanded from the ground.

7.2.4.4 Shunt Limiter

The shunt regulator makes use of a proportional shunt and several switching shunt elements. Since there is a single proportional shunt element and its failure could result in battery damage or solar power degradation, it is implemented using a quad configuration which gives protection against all single part failures and the majority of double part failures. The switching shunt elements could fail, again resulting in solar array power degradation or loss of the main bus limiting function. Since the array output is reducing as Mars is approached, there is only a relatively short period of time when all shunting elements are required. If a switching shunt element should fail open, then a chance is taken that the main bus might experience a temporary overvoltage. This could be minimized by applying extra power loads to the bus by ground command. If a short occurred, loss of that section of the array shunted by that element would be occasioned. The design of the switching shunt element assemblies therefore makes use of series redundancy to ensure that protection against short circuit failure modes is provided.

7.2.4.5 Synchronization Signal Generator

This element provides a signal to the telemetry, computer, and sequencer functions, and failure could compromise the mission. The signal generator is therefore designed to be function-redundant so that failure of the primary unit causes a redundant unit to be switched in.

7.2.4.6 Logic Circuits

The logic circuits required to implement correct selection of line voltage limiter, battery charge rate, and priority load switching are designed to be either part redundant or to function in a majority voting logic fashion. Under these ground rules no single part failure can cause degradation of performance to an unacceptable level. This same design concept is utilized in the implementation of the command processing circuitry. The command circuitry itself is included to provide ground-commanded control of battery charging, and reconditioning as a further means, in conjunction with telemetry signals, of improving the reliability of the battery operation.

7.2.4.7 400-Hz Inverter

Failure of this unit would cause severe mission degradation since it supplies power to the antenna and PSP drives. It is designed to be function-redundant and to be automatically switched from the main inverter to the redundant standby inverter should a failure occur.

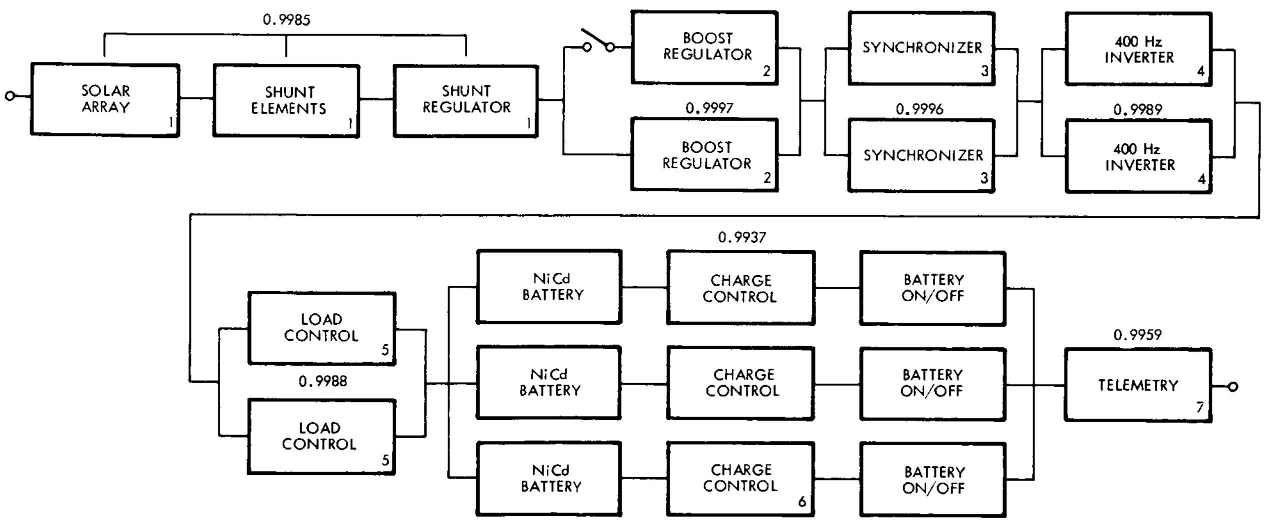
7.2.4.8 DC-DC Converters

These units condition the main bus power to the appropriate voltage, regulation, and power levels of the various user loads. The failure protection requirements for each of the units and the mission criticality of the loads must be reviewed prior to establishing the total redundancy concept for each of the converters. Redundancy can be implemented at the part or circuit level depending upon requirements and will be so incorporated when requirements are fully defined.

7.2.4.9 Reliability Assessment

A reliability assessment has been made for the electric power subsystem based on the anticipated parts count and circuitry design concepts for the various subsystem elements. Following is a compliation of the assessment. Figure 7-9 illustrates this calculation.

Subsystem Element	Probability of Success
Solar array and shunts	0.9985
Boost regulator	0.9997
Synchronizing signal generator	0.9996
400-Hz inverter	0.9989
Load control logic circuits	0.9988
Battery and control circuits	0.9937
Subsystem overall	0.985



$$R_{ELECTRICAL\ POWER} = R_1 \left[R_2(1+\lambda_2 T)\right] \left[R_3(1+\lambda_3 T)\right] \left[R_4(1+\lambda_4 T)\right] \left[R_5(1+\lambda_5 T)\right] \left[R_6^3 + 3R_6^2(1-R_6)\right] R_7$$
$$= 0.9850$$

Figure 7-9

RELIABILITY OF ELECTRICAL POWER SUBSYSTEM is estimated as 0.985 on the basis of the above block diagram. Detailed calculations are presented in Volume 2, Section 3.

7.3 COMPONENT DESCRIPTION

7.3.1 Solar Array

The solar array is formed by mounting solar cells on the bottom of the equipment module and on a rigid annulus around that module (Figure 7-10). The array is thus completely fixed and requires no deployment mechanism.

Because the panels at the bottom of the equipment compartment can dispose of heat only through their front surface, they are hotter than the panels on the annulus, which can also radiate heat through their back surface. There are, therefore, two distinct arrays operating at different temperatures but feeding into the same power bus. To obtain the same voltage out of each of these two arrays, each has a different number of cells connected in series.

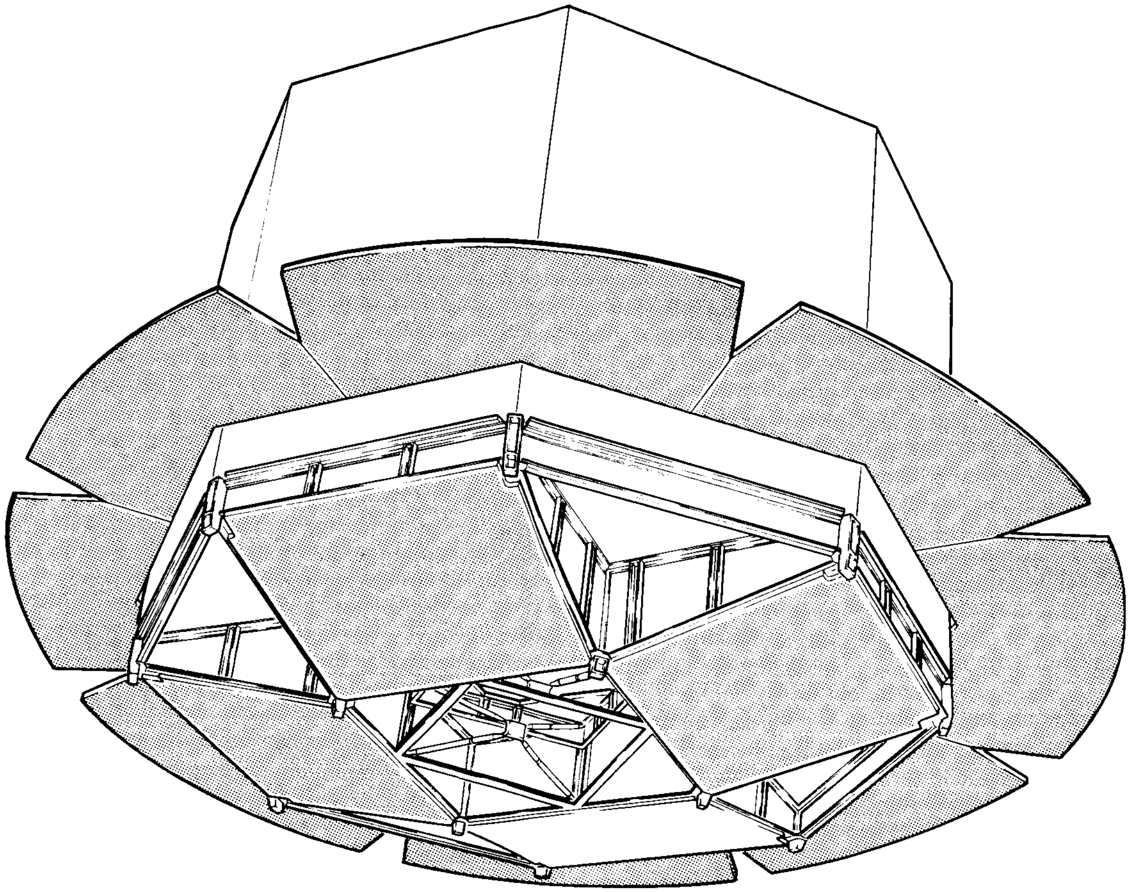
The type of cells, cover slides (Table 7-3) and modules (Figure 7-11) recommended for Voyager spacecraft are presently used on INTELSAT III and a classified program. It is the so-called overlapped cell design. The module has several advanced features desirable for a Mars orbiter. The cells are connected with flexible straps capable of taking severe thermal cycles from -170 to +125°C without damage. Kovar and molybdenum connecting bars have been tested with less than one-percent damaged Kovar connectors, after numerous temperature cycles, and no damaged molybdenum connectors. The cells are overlapped to maximize the active solar cell area. Fifteen- and 18-cell modules, as shown in Figure 7-11, are used in the layout. The insulated panels use the 18-cell module exclusively (20 of them), while the uninsulated panels use 15 18-cell modules and four 15-cell modules.

The characteristics of these arrays are shown in Figure 7-12 (the insulated array) and Figure 7-13 (the uninsulated array), with typical layouts of the cell modules in Figure 7-14 and 7-15.

The array degradation factors listed in Table 7-4 are taken from References 1 and 2. A two-percent loss was assumed for the array sterilization as a safety margin. However, no degradation due to sterilization is expected.

PRELIMINARY SPECIFICATION

Solar Array



Purpose	**Performance Characteristics**	
The solar array fits well within the alunch vehicle shroud and provides growth potential of about 300 watts at 1.62 AU by filling empty panel areas.	Power output at 1 AU	1426 watts
	Power output at 1.67 AU	787 watts
	Total area	226 sq ft
	Total weight	217 lb
	Voltage at 1.67 AU	52.0 volts
	Cell type	N- on-P, 2x2 cm 7 to 14 ohm-cm

The solar array of the recommended spacecraft consists of an annular section and three panels on the base of the equipment module. No deployable or moving panels are used. Space is available on the base of the equipment module for further solar cells if spacecraft performance is to be upgraded.

Figure 7-10

Table 7-3. Solar Cell and Cover Slide Characteristics

Solar Cell:	Material: Silicon
	Type: Non P (TiAg sintered contacts)
	Size: 2 x 2 cm
	Thickness: 0.015 in.
	Efficiency: 10.5% (Air mass zero, 28°C)
	Basic Resistivity: 7-14 ohm-cm
	Series Resistance: 0.5 ohms
	I_{sc} = 138.0 ma
	V_{oc} = 0.550 volts
	I_{op} = 126.5 ma
	V_{op} = 0.440 volts
	$\beta V_{oc} = \begin{cases} 100^\circ C = -2.25\ mV/^\circ C \\ 60^\circ C = -2.20\ mV/^\circ C \\ 0^\circ C = -2.15\ mV/^\circ C \\ -30^\circ C = -2.05\ mV/^\circ C \end{cases}$
Cover Slide:	Material: Fused Silica
	Size: 1.8 x 2 cm
	Thickness: 0.006 inch
	Cut-off Frequency: 410μ
Cell to Cell Connectors:	Kovar or molybdenum

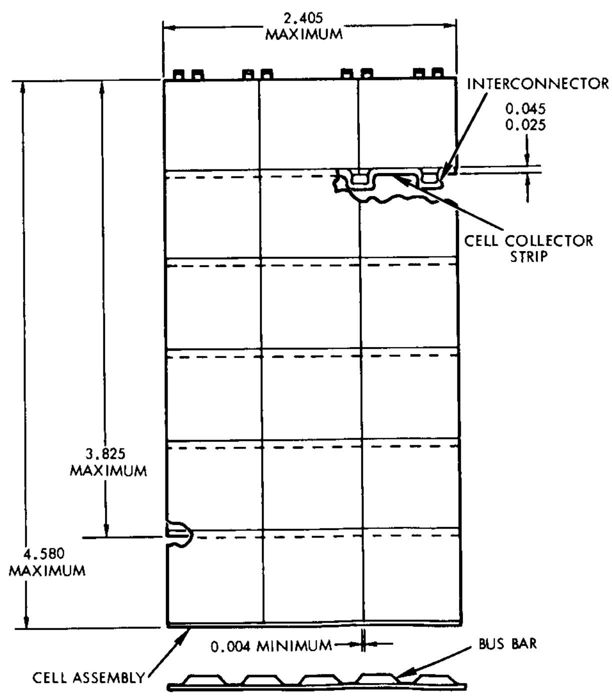
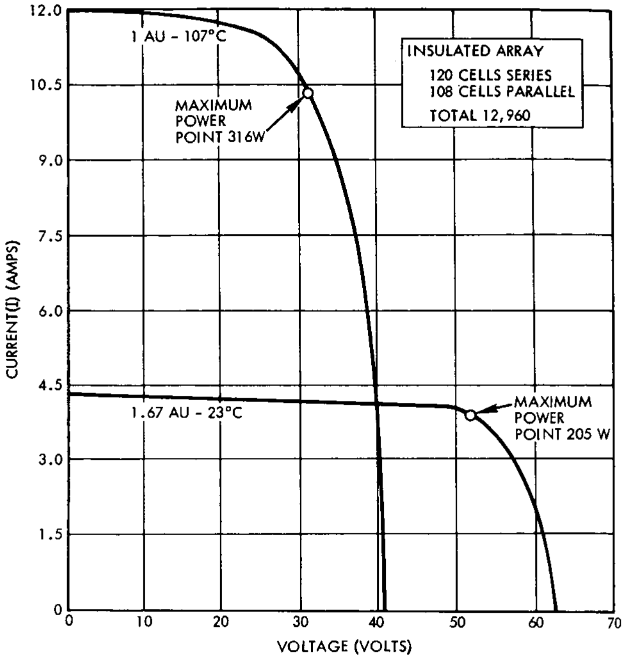


Figure 7-11

FLEXIBLE 15 AND 18 CELL MODULES, incorporating advanced techniques developed by TRW, withstand thermal cycles from - 175°C to +125°C. The slight overlaps of the cells increase the active area (95% of the cell area with overlapped modules compared to <88% for flat cells).



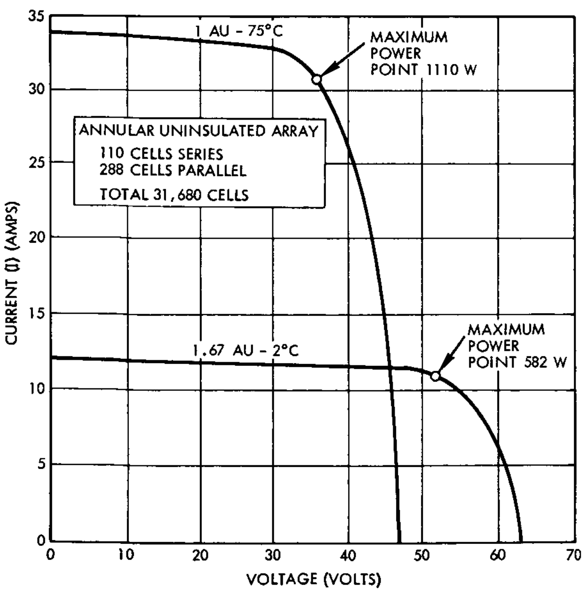
INSULATED PANELS

TOTAL AREA	61 SQUARE FEET
CELLS IN SERIES	120
CELLS IN PARALLEL	108
TOTAL CELLS	12,960
PACKING FACTOR (CELL AREA/TOTAL AREA)	91%
WEIGHT	49.6 POUNDS

	AT 1 AU	AT 1.62 AU	AT 1.67 AU
OPERATING TEMPERATURES °C	107.0	28.0	23.0
POWER OUTPUT, W	316.0	220.0	205.0
POWER OUTPUT PER UNIT AREA WATTS/SQUARE FOOT	5.19	3.6	3.36
VOLTAGE AT MAXIMUM POWER POINT	30.5	50.0	52.0

Figure 7-12

INSULATED SOLAR ARRAY CHARACTERISTICS show the effect of operating temperatures on the current- voltage curves. The high temperature occurs near earth, the low temperature near Mars. The thermal insulation for this array is provided by the base of the equipment module, on which it is mounted.



UNINSULATED PANELS

TOTAL AREA	165 SQUARE FEET
CELLS IN SERIES	110
CELLS IN PARALLEL	288
TOTAL CELLS	31,680
PACKING FACTOR (CELL AREA/TOTAL AREA)	83%
WEIGHT	167 POUNDS

	AT 1 AU	AT 1.62 AU	AT 1.67 AU
OPERATING TEMPERATURES °C	75	7.0	2.0
POWER OUTPUT, W	1110	616.0	582.0
POWER OUTPUT PER UNIT AREA WATTS/SQUARE FOOT	5.74	3.74	3.53
VOLTAGE AT MAXIMUM POWER POINT	36.0	50.0	52.0

Figure 7-13

UNINSULATED SOLAR ARRAY CHARACTERISTICS show the temperature extremes between near earth operation and orbital operation at Mars. The uninsulated array is located on the annular ring around the base of the equipment module.

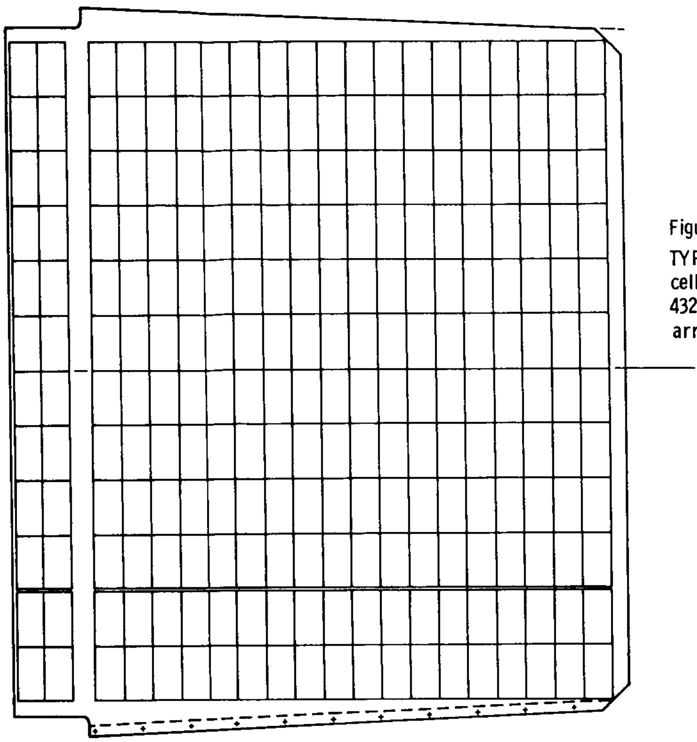


Figure 7-14
TYPICAL RECTANGULAR PANEL shows the layout of twelve 120 series cells strings in 18 cell modules (3 cells in parallel), for a total of 4320 cells in each panel. The three panels comprising the insulated array have the characteristics shown in the current-voltage curve,

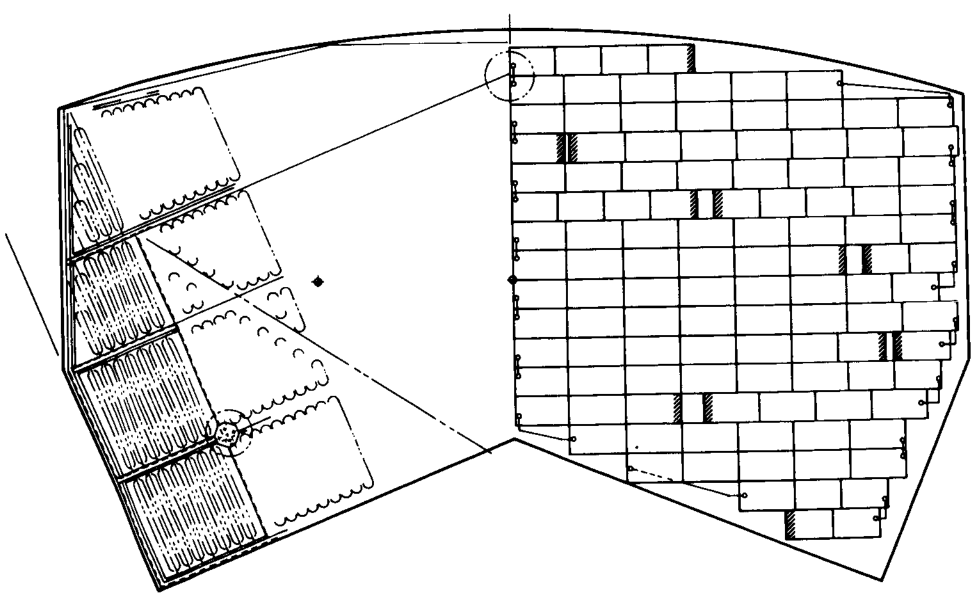


Figure 7-15
TYPICAL ANNULAR PANEL, uninsulated, shows the layout of 15 and 18 cell modules, 12 strings with 110 cells in series and three in parallel can be installed in each panel for a total of 3960 cells. The eight panels comprising the annular Array have the characteristics shown in the current-voltage curve, Figure 7-13.

Table 7-4. Degradation Factors

Degradation Factors:
(After one year)*

	I(%)	V(%)
Assembly losses	1	1
Losses due to cover slide	4	
Array sterilization	1	1
Solar flares*	3	
Meteorite sputtering*	2	
UV irradiation*	5	
Total Losses*	16	2

The loss assumed for solar flares is an estimate. The probability of no solar flare encounters for a one-year period is 0.95 (Reference 3). If solar flares should occur, the magnitude should be proportional to $1/AU^2$. Diode and wiring losses are included in the power requirement and all assumed to be two volts.

7.3.2 Batteries

The preliminary specifications sheet for the battery is given in Figure 7-16. The battery tradeoff studies that resulted in the selection of NiCd batteries are discussed in Appendix Q.

7.3.3 Power Control Unit

The configuration of Voyager power control unit is shown in the subsystem block diagram (Figure 7-4). A preliminary specification sheet is shown in Figure 7-17. The PCU consists of the boost regulator with standby redundancy, the redundant sequential shunt regulator, the synchronizer with standby redundancy, power switching logic, battery charge control logic, command capability, and telemetry monitors. It is 6 by 10 by 6 inches and weighs 26 pounds. The tradeoff studies of various regulator approaches are given in Appendix R. The list of commands is shown in the command list in Section 3. The telemetry list for monitoring the power subsystem is part of the spacecraft list shown in Section 1.4.

PRELIMINARY SPECIFICATION
Battery

Performance Characteristics
(single battery)

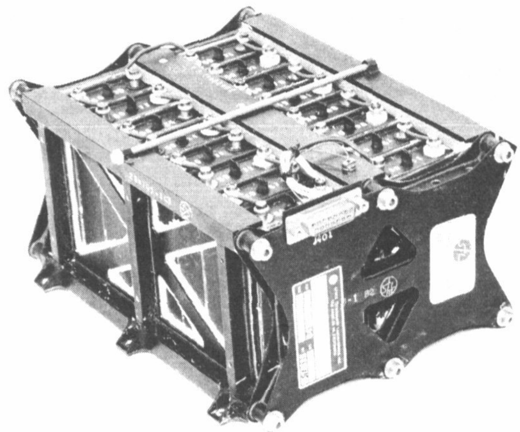
1.	Nominal capacity:	16 ampere hours
2.	Voltage range:	37 to 50 volts
3.	Average discharge current:	3.5 amperes
4.	Nominal discharge voltage:	41 volts

Physical Characteristics

1.	Battery type:	Hermetically sealed nickel-cadmium
2.	Number of cells:	33 series connected
3.	Charge control sensing:	Voltage (temperature compensated) cutoff
4.	Weight:	50 pounds
5.	Size:	16.9 x 8.6 x 6.7 in.

Battery System
(3 batteries)

1.	Nominal capacity:	48 ampere hours
2.	Depth-of-discharge:	50 percent
3.	Weight:	150 pounds



TRW NICKEL-CADMIUM BATTERY (TYPICAL) USED ON VELA

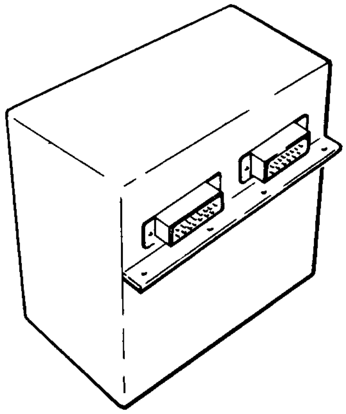
Figure 7-16

7.3.3.1 Boost Regulator

A block diagram of a boost regulator is shown in Figure 7-18. The regulator is controlled by a proportional signal from the error amplifier of the main bus voltage sensor. This regulator consists of a constant-frequency pulse-width modulator which controls a power inverter. The power inverter has, as its input, solar array voltage. This voltage is stepped up in the inverter and the resulting AC output rectified, filtered, and supplied to the main bus. Since the bus voltage will be clamped to the battery voltage prior to switching the battery to trickle charge, the boost regulator will be boosting the main bus voltage toward the voltage limit, 50 volts ±1 percent DC. It will reach this limit and maintain it when the last battery has reached its temperature-compensated voltage limit. A current limit also controls the boost regulator to preclude charging a battery at too high a current rate. The major protion of the main bus power is supplied directly through a series diode in the boost regulator. The inverter therefore handles only the additional power needed to raise

PRELIMINARY SPECIFICATION

Power Control Unit and Shunt Assembly



WEIGHT 20 LB
DIMENSION 11 x 5.5 x 3.5

POWER CONTROL UNIT

WEIGHT 6 LB
DIMENSION 7.5 x 6.5 x 5.5

SUNT ASSEMBLY

Purpose

Output voltage 37 to 50 VDC

Solar Array control
  Limit solar voltage to 50 VDC ± 1%

Battery Control
  Provide battery charge control
  Provide battery discharge control
  Provide battery disconnect

Load Control
  Provide nonessential load control

  Connect capsule load when normal load is connected, batteries are not charging, and excess array capacity exceeds 200 watts.

  Connect RCS gas heater load when capsule load is connected and excess array capacity exceeds 128 watts.

  Disconnect heater load when batteries discharge

  Disconnect capsule load when heater load is disconnected and batteries discharge.

Synchronizer
  Provide sync for spacecraft equipment:
    819.2 kHz
    4 kHz
    400 Hz

Telemetry
  Provide telemetry signals: Main bus voltage
    Total current
    Solar array temperature
    Battery temperatures

Figure 7-17

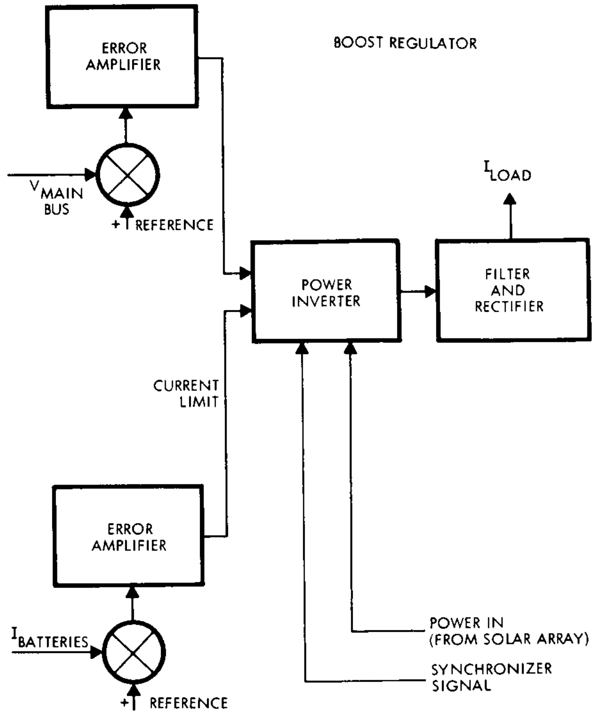


Figure 7-18
BOOST REGULATOR increases Solar Array voltage up to a controlled limit of main bus 50 VDC ±1%.

the array voltage to the required output level. The ratio of the output-to-input voltage is the same as the ratio of total drive period to the OFF time of the switching element. Efficiencies of 95-percent have been obtained using this scheme, becoming greater as the input/output voltage ratio approaches unity. The regulation limit and cycle for the boost regulator is apparent in Figure 7-7. Synchronization of the boost regulator is obtained from the synchronizer in the PCU. An identical boost regulator in the standby mode is used to obtain the desired reliability. Logic will be provided to accomplish the switching of circuits. The power handling capability of the boost regulator is about 1500 watts.

7.3.3.2 <u>Shunt Regulator</u>

The shunt-regulator (Figure 7-19) is a combination tapped-array, dissipative shunt control and sequentially switched array control. When the array capacity exceeds the load requirements, an error signal will be generated by the voltage sensor circuit. This error signal goes to a linear amplifier which controls the current of the linear

SEQUENTIAL SHUNT REGULATOR

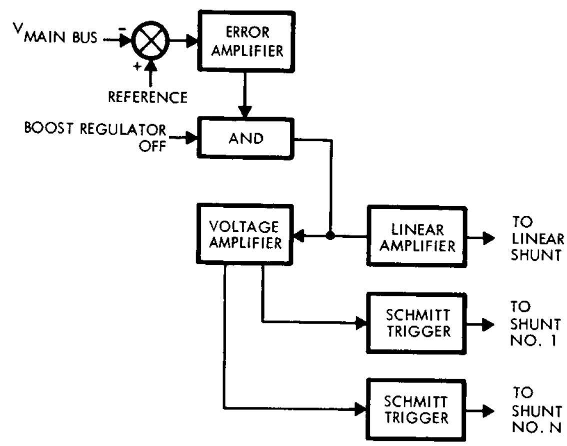


Figure 7-19
SEQUENTIAL SHUNT REGULATOR limits main bus voltage when Boost Regulator is off, using sequential shunts to minimize thermal dissipation.

shunt element. The range of control of the linear shunt element has been tentatively selected to be from 0 to 5 amps. The same error signal to the linear amplifier is also sent to a constant gain voltage amplifier, where it is amplified to provide an expanded analog of the error control signal. The constant gain voltage amplifier provides the signal which will determine how many additional shunt elements must be saturated in order to maintain the required regulation limits. The signal is fed to a group of parallel level detectors, Schmitt trigger circuits. The level detectors are designed so that each turns on at a predetermined voltage, thus saturating the appropriate shunt element. Sequential operation of the shunt elements is ensured because the "on" threshold of a circuit is set at a higher voltage than the preceding circuit. Wide separation and stability of the turn-on levels is obtained through the constant-gain voltage amplifier. The input to the amplifier would be the error signal generated by the voltage sensor.

When the load requirement is exactly equal to the solar array capacity, all shunts will be off. As the array capacity increases or the load decreases, the output of the error amplifier increases proportionally, turning on the linear amplifier and its shunt element. The threshold voltage of the first on-off shunt element is set so that it saturates if the array capacity is greater than the load requirement plus the shunt current of the linear element. The remaining shunt elements turn on at each successive two amp increment. The current shunting capacity of one linear amplifier (five amps) and eight power switches (two amps each) is 21 amps. The maximum short-circuit current of the proposed Voyager solar array at Mars encounter is less than 20 amps. Since all shunts monitor the error signal simultaneously, the speed of response to load and solar array variations is quite satisfactory.

The power dissipation of the linear amplifier shunt stage is approximately 50 to 100 watts. The other shunt elements dissipate about four watts each. This scheme therefore limits main bus voltage by controlling the array current and minimizes the area required for shunt heat dissipation. The regulation limit and cycle for the shunt regulator is shown in Figure 7-7. Redundancy can be obtained by quad, series, and majority voting techniques.

7.3.3.3 Synchronizing Signal Generator

Synchronization for the entire spacecraft is provided from redundant precision oscillators (819.2 kHz) located in the electric power subsystem.

A 819.2 kHz crystal with a 0.01-percent tolerance establishes the reference clock. Therefore, the maximum end-of-life frequency drift can be 80 Hz. Each oscillator is driven by a crystal-generating the 819.2-kHz master clock rate. This master clock rate is used directly by the C and S subsystem. Through a series of mod counters, this frequency is divided into the appropriate frequency required by the various subsystems. The 819.2-kHz master clock signal is also divided by a modulo 205 counter to drive the 4-kHz clock, and a mod 10 counter is driven by the 4-kHz clock to generate the 400-Hz clock. Two redundant synchronization assemblies are provided, only one of which is operative at any time (see Figure 7-20). A narrow tuned filter senses the frequency of operation. Under normal operating conditions the proper frequency produces a signal to keep the relay closed such that Channel A is in operation. In event of a failure of any portion of Channel A which produces an improper output frequency or voltage level, the relay will drop out and Channel B will be used to operate the system.

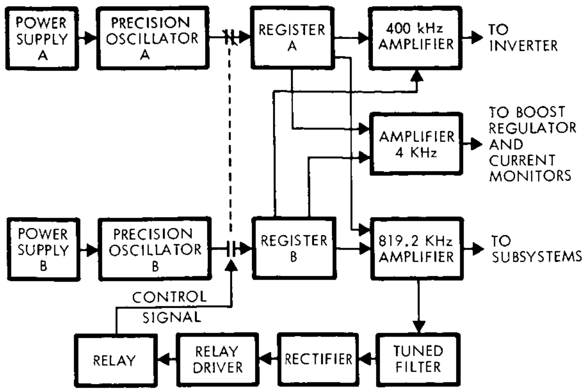


FIGURE 7-20

REDUNDANT PRECISION SYNCHRONIZER provides 400 Hz to Inverter, 819.2 kHz to the spacecraft subsystems, and 4kHz to booster regulator current monitors and power conditioning equipment. 4kHz is selected to have good efficiency in switching transistors in the regulator and the DC/DC converters.

The countdown register consists of micrologic binary units. The amplifiers and relay driver are quad redundant for high reliability.

7.3.4 400-Hz Inverter

The preliminary specifications for the 400-Hz inverter appear in Figure 7-21.

7.3.5 Main Bus

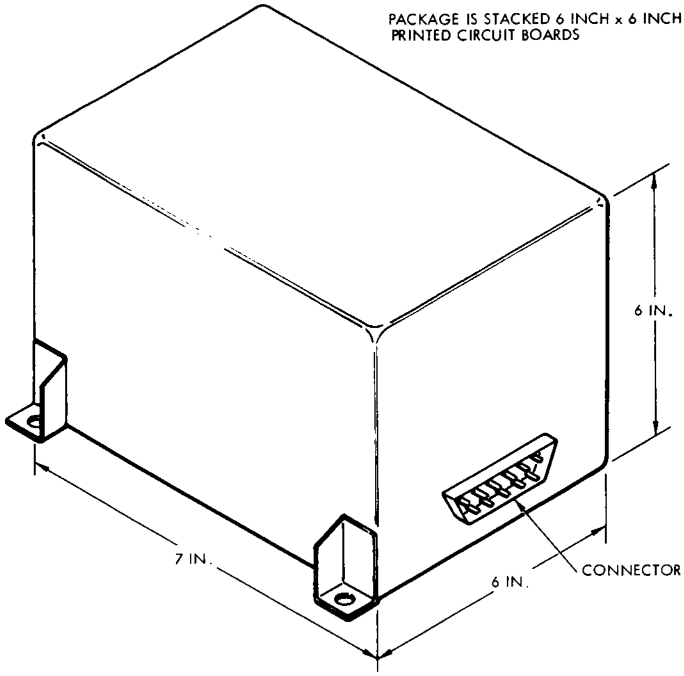
The main bus, through which all raw power is delivered from the power subsystem, will be regulated by the characteristics of the battery and will have an upper limit value determined either by the boost regulator or the sequential shunt limiter. As a result, the bus voltage will be in the range of 37 to 50 VDC. The ripple, noise, EMI, and impedance characteristics are not presently specified and will require definition at a later date.

7.3.6 DC-to-DC Converters

The DC-to-DC converters will be designed to supply power at the voltage and quality required by the specific loads. Decentralized units, each one serving a single subsystem, will have several regulated outputs since loads for each subsystem will have different voltage needs. The

PRELIMINARY SPECIFICATION

400 Hz Sine Wave Inverter



Physical Characteristics

Size: 6 in. w x 7 in. d x 6 in. ht (250 $in.^3$)
Weight: 10.0 lb
Effeciency: 85%
Input voltage: 37-50 VDC
Output voltage: 50 V RMS sine wave, 2ϕ, 0° and 90°
Output voltage regulation: ±2%
Frequency stability: Synchronized, ±0.01%
Unsynchronized, ±5%
Output power: 50 VA total
Redundance: Standby with low voltage sensing and automatic channel switching

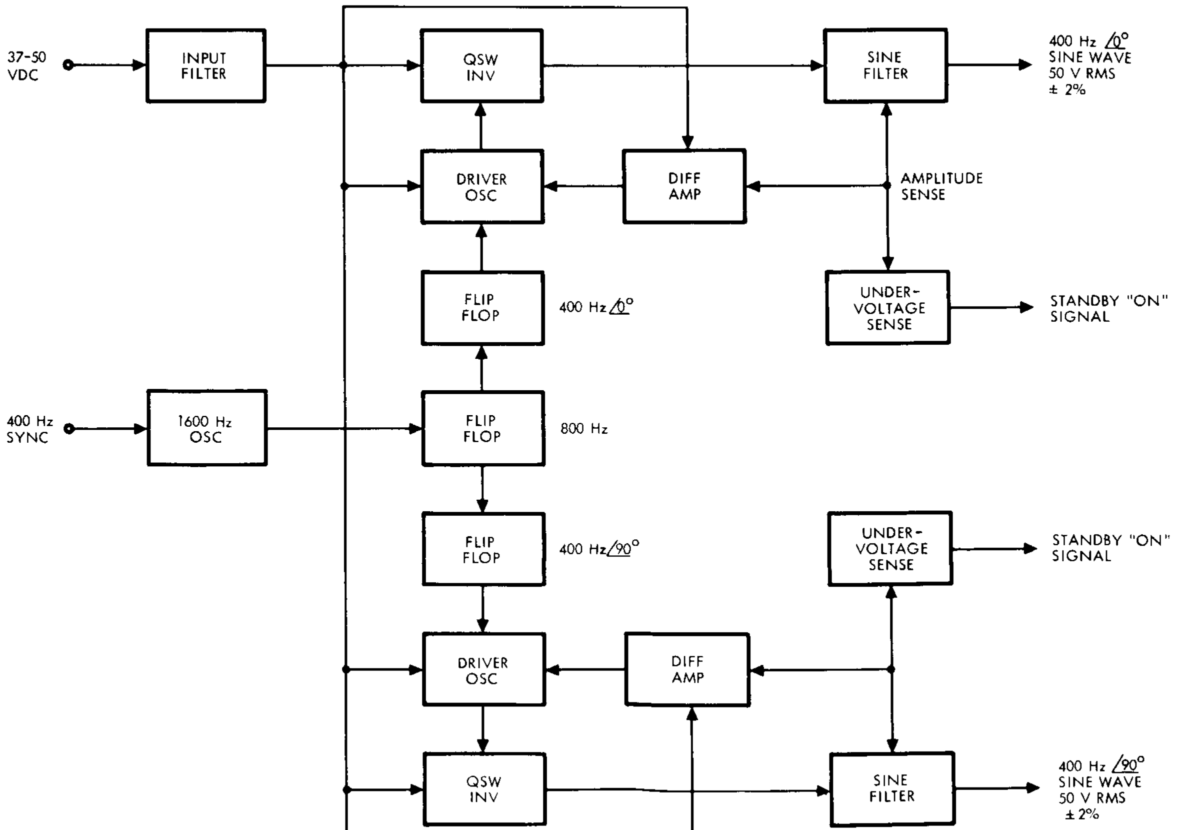


Figure 7-21

S-band radio subsystem will probably require +15, -15, and +28 VDC. These will all be supplied from a single unit. Each unit is optimized for power utilization efficiency after the requirements are specified. Computer programs for this operation are being developed by TRW under NASA-sponsored programs: "Study and Analysis of Satellite Power Systems Configurations for Maximum Utilization of Power" for Goddard Space Flight Center, and "Power System Configuration Study and Reliability Analysis" for Jet Propulsion Laboratory.

8. ELECTRICAL DISTRIBUTION AND PYROTECHNIC CONTROL

8.1 SUMMARY

The electrical distribution and pyrotechnic control subsystem electrically interconnects the various elements of the planetary vehicle and launch vehicle by use of harnesses, junction boxes, and umbilical connectors. It also provides firing control for all ordnance devices. A distribution control unit within the subsystem acts as a source distribu-point for primary DC power to all other subsystems.

The subsystem preliminary specification in Figure 8-1 shows principal subsystem parameters and design features. An outline of the subsystem on the spacecraft is sketched in Figure 8-2.

All techniques of equipment design in the electrical distribution and pyrotechnic control subsystem are based upon use of components and methods for which a great deal of TRW experience has been accumulated on such programs as the Orbiting Geophysical Observatory, the Vela Nuclear Detection Satellites, Pioneer, INTELSAT III, and others.

8.1.1 Constraints

The electrical cables to the planetary scan platform accommodate ±180 degrees of rotational motion of the platform about each of two axes.

The pyrotechnic control subsystem and initiator cartridges must comply with sterilization requirements.

Chief pyrotechnic initiator requirements are:

- The probability of successful operation of the cartridge under Voyager preflight and flight environments will not be less than 0.996 when supplied with the all-fire current of 4.5 amperes.
- Cartridges will perform satisfactorily at any temperature in the range of -65 to +250°F.
- Electropyrotechnic cartridges will conform to AFETRM 127-1 Range Safety Manual for category "A" devices.

PRELIMINARY SPECIFICATION

Electrical Distribution and Pyrotechnic Control Subsystem

Purpose

To provide electrical interconnection at frequencies below 2 MHz for the various elements of the flight spacecraft and the launch vehicle and capsule, and to provide safe/arming and firing circuitry for spacecraft electro-explosive devices.

Type

Proven lightweight harnesses, J-boxes, and electronic assemblies. Hot bridge-wire initiators with relay control. Single bridge-wire devices, battery initiation. Redundant pyrotechnic control system and redundant initiators.

Performance Characteristics

Interconnects elements of flight spacecraft.
Interconnects spacecraft, launch vehicle, and capsule.
Provides systems test direct wire access to spacecraft.
Provides prelaunch direct wire access to spacecraft.
Provides a source distribution point for all primary power.
Provides a safe/arm function for all ordnance devices.
Provides firing control signals for all ordnance devices.
Complies with Range Safety Document AFETRM 127-1

Physical Characteristics

Separate equipment module and propulsion module J-boxes (7 each).

Functionally separated harnesses in equipment module, propulsion module and PVA.

External harnesses protected against environment.

Weight: 306 pounds

Reliability 0.996

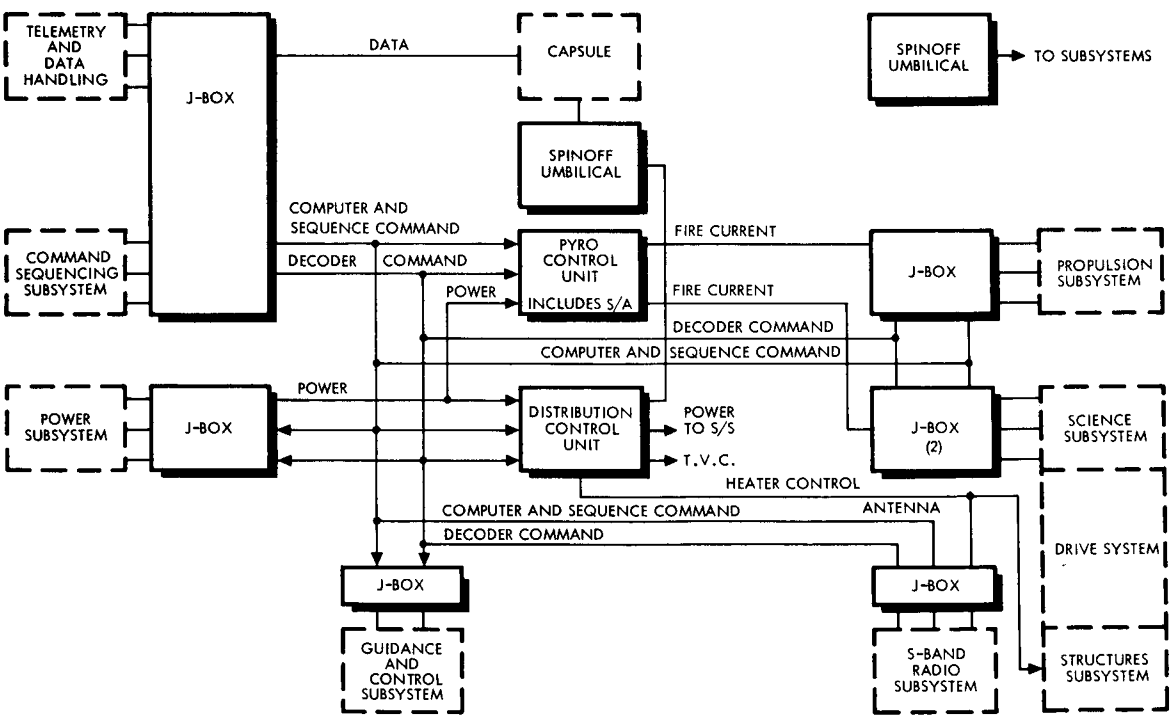


Figure 8-1

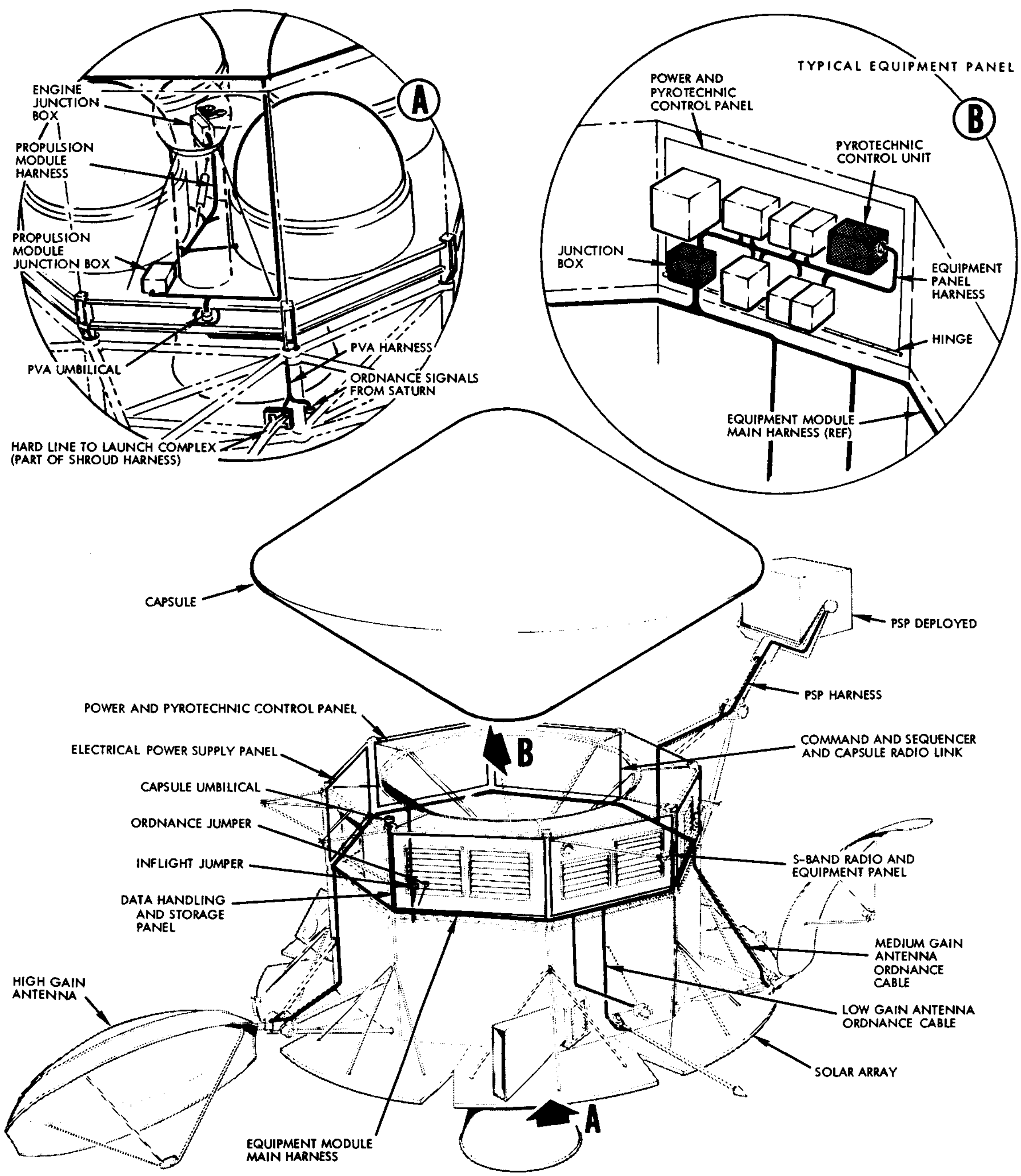


Figure 8-2

THE ELECTRICAL DISTRIBUTION AND PYROTECHNIC CONTROL SUBSYSTEM interconnects the subsystems on the equipment module, power module and launch vehicle adaptor.

- Electropyrotechnic cartridges will be hermetically sealed.
- Flight cartridges will be qualified by TRW Systems by lot qualification testing. This process is described in more detail in the Implementation Plan.

8.2 SUBSYSTEM FUNCTIONAL DESCRIPTION

The block diagram of the electrical distribution and pyrotechnic controls subsystem is shown in Figure 8-3.

8.2.1 Harnesses and Junction Boxes

The harnesses and junction boxes are designed to cause minimum voltage drop which, as a design goal, will be less than two percent of the source voltage.

8.2.1.1 Harnesses

It is planned that the majority of the harnesses will be designed only once and the same configuration will be usable in several different flight spacecraft. The science harnesses will be designed for a specific set of scientific experiments as required for a given mission and would be redesigned for any spacecraft having changed experiments. This same concept is applicable to junction boxes and has been very successfully used on large spacecraft previously delivered by TRW Systems.

Redundant circuit wiring throughout is not used, however, circuits that are themselves redundant are served by separate wires, thus making the system redundant. Figure 8-4 illustrates this concept.

A shroud/spacecraft umbilical connector provides prelaunch charging power for the batteries in the spacecraft and monitoring lines as required. This connector is separated prior to spacecraft/launch vehicle separation. The external cable between the shroud and ground control facilities is removed prior to liftoff.

A capsule/spacecraft umbilical connector allows the monitoring of certain capsule functions in the spacecraft and supplies capsule power as required. The ejection initiation signal will be supplied to the capsule through the umbilical.

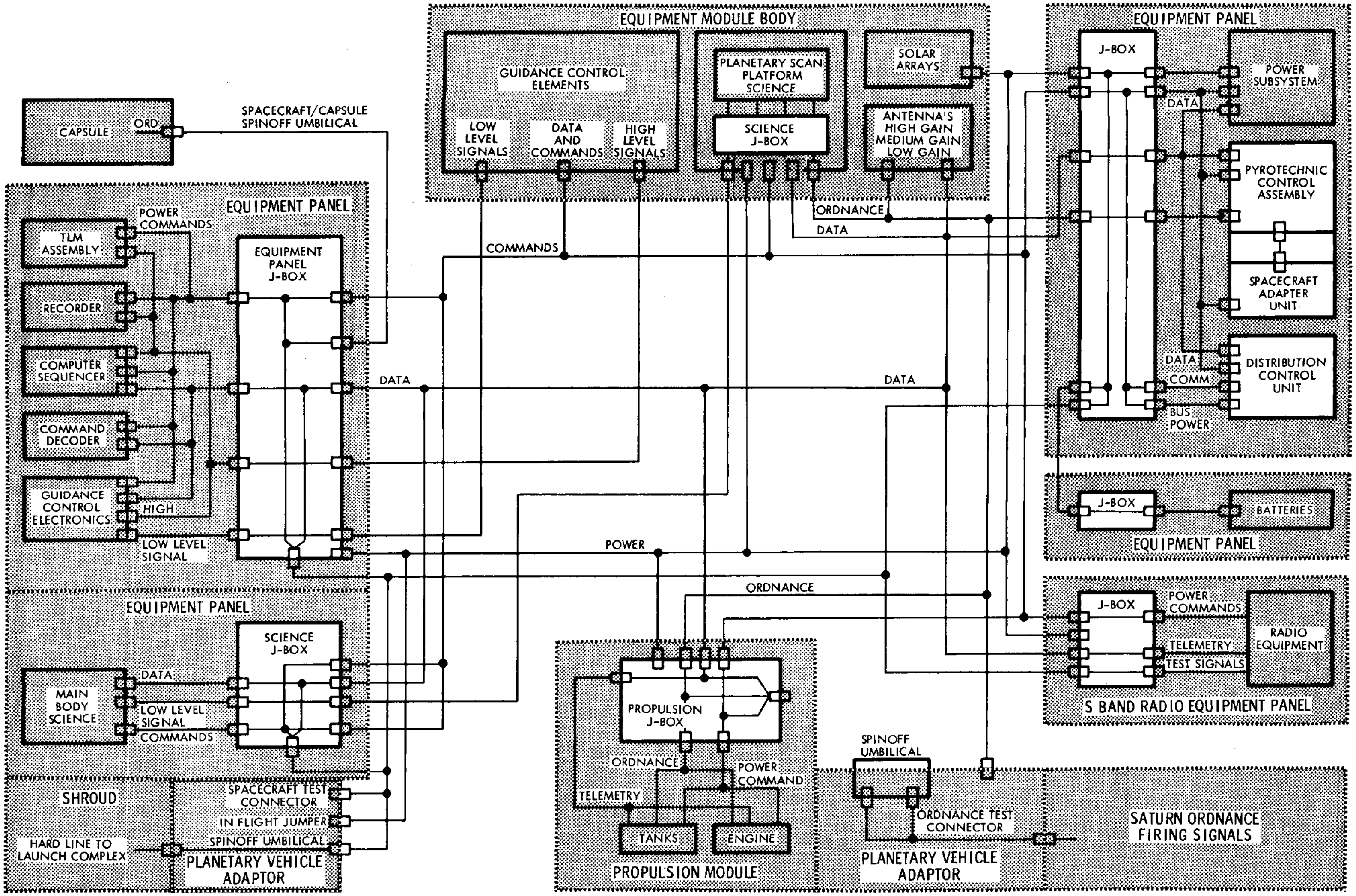


Figure 8-3

THE ELECTRICAL DISTRIBUTION AND PYROTECHNIC CONTROL SUBSYSTEM interconnects the spacecraft subsystems by means of separable harnesses and junction boxes. The pyrotechnic control assembly and the distribution control unit are contained within this subsystem.

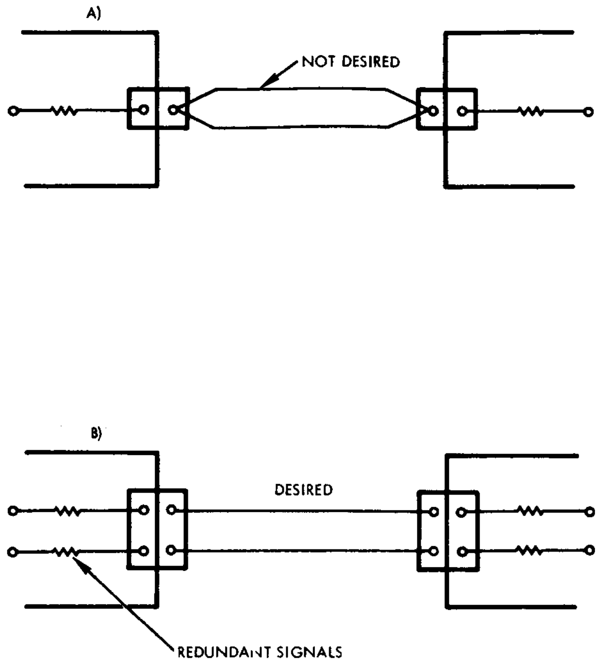


Figure 8-4
REDUNDANT WIRING CONCEPT applied in the recommended spacecraft uses completely redundant circuits rather than duplicating cables between nonredundant circuits.

An in-flight jumper connector provides a bridge across a break in each primary power bus circuit to allow the spacecraft to function. The connector is pulled out when all spacecraft buses must be de-energized. During system test operations individual bus power measurements can be made at this connector interface and also individual buses can be switched on and off by a simulator. This technique has been used on many other spacecraft programs at TRW. Figure 8-5 illustrates this concept.

An ordnance jumper connector is used for ordnance system testing and shorting. This connector is wired to circuit points within the pyro control unit and will be accessible during spacecraft integration and test operations. A sketch illustrating this ordnance test connector is shown in Figure 8-6.

8.2.1.2 Standard Junction Boxes

Junction boxes supply protected mounting locations for electrical branch points (terminals) and electrical components (resistors, capacitors, diodes, etc.). They also aid in establishing the concept of separable

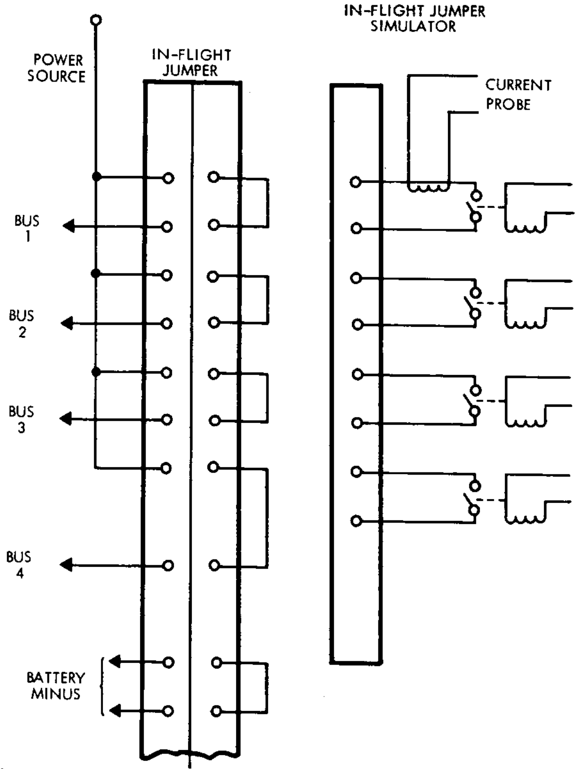


Figure 8-5
THE IN-FLIGHT JUMPER provides a means of removing power from spacecraft subsystems during ground testing. Testing capability is enhanced by the addition of the in-flight jumper simulator.

harnesses. Several harnesses can interface with each other by use of a junction box. Each harness serves a specific category of equipment and thus can be identified as an individual functional assembly. The junction box allows harnesses to be kept to a reasonable size by interconnecting several harnesses.

Junction boxes also solve the problem of signal and power distribution throughout the spacecraft. Often a single function has to be delivered to several destinations, which means that a multiple branch point must be established somewhere from which power can be distributed. The junction box provides this function. In cases where only a break in harnesses is required, the junction box shrinks to a simple bracket acting as a connector interface. Subsystem junction boxes are "universal", which means they serve both the recommended spacecraft and the upgraded version.

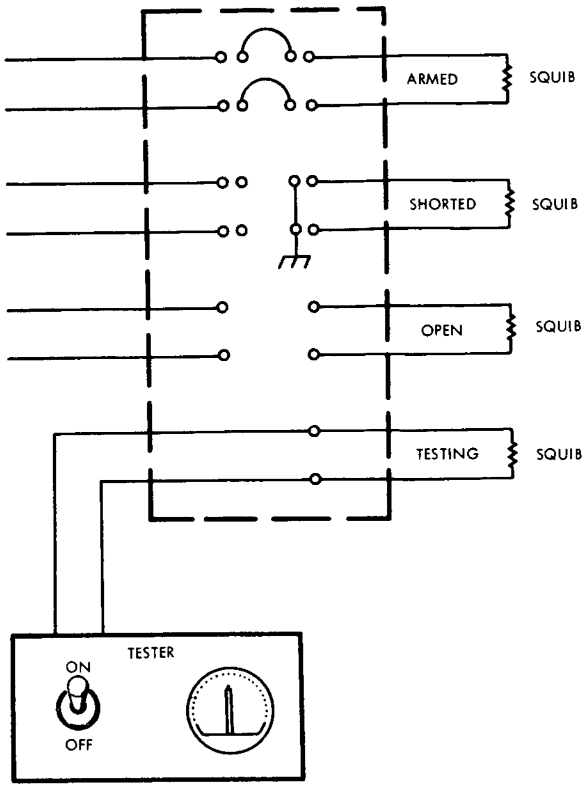


Figure 8-6
THE ORDNANCE JUMPER CONCEPT recommended permits testing of squibs during the prelaunch period.

8.2.1.3 Science Junction Boxes

The science junction box establishes a location for the interfaces between experiments and spacecraft subsystems, and between experiment equipment in one area and experiment equipment in other areas. These assemblies are the gathering places for all subsystem signals routed to experiments and all subsystem input points receiving signals from experiments. Figure 8-7 illustrates the typical science junction box concept. These junction boxes cannot be universal and will be redesigned for each changed set of experiments.

8.2.1.4 EMC Considerations

To provide effective interference shielding for its internal circuitry, the spacecraft junction boxes are designed to be radio-frequency tight and electrically bonded to the spacecraft conductive metal structure. Each enclosure is electrically continuous with high conductivity across each seam, joint, or other discontinuity. Very small openings or gaps are

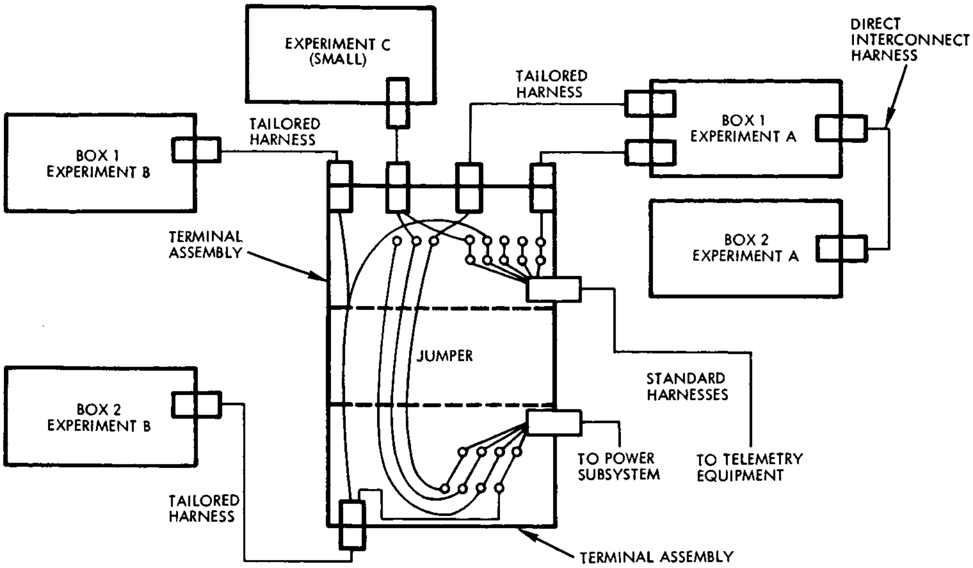


Figure 8-7

THE SCIENCE JUNCTION BOX enables the simple accommodation of different science payloads for changing mission objectives.

allowable. The enclosure and its receptacles are finished with electrically conducting finishes such as gold iridite and gold. Enclosure-to-structure bond impedances will be 2.5 milliohms DC or less.

The internal packaging of the electrical distribution subsystem units is designed to minimize the coupling of interference between circuits. Several interference control measures employed are:

- Locating power switching relays in shielded compartments and decoupling the contact circuits with bulkhead mounted, feedthrough filters
- Twisting and shielding all circuits which generate interference or are susceptible to interference
- Grounding wire shields at each end to maximize their shielding efficiency. Bundling interference sensitive wiring going to interference sensitive spacecraft equipment separately from noisy wiring
- Routing any AC power, primary DC power, secondary DC power, or commands on separate output connectors to avoid coupling; in passing through these connectors, carry each two-wire circuit on adjacent pins to minimize the circuit area and, in turn, the interference pickup or generation

- Twisting and shielding all two-wire circuits which generate, or are susceptible to, interference. The shields will be multiple point grounded directly to the spacecraft conductive metal structure.

- Bundling primary power, secondary power, digital signal, and other signal wires separately to effect isolation between generating and susceptible circuits. To further reduce coupling, these bundles will be physically separated from each other as far as practicable.

- Routing of wiring harnesses as close as possible to the spacecraft conductive metal structure to minimize generation or pickup of electromagnetic fields

- Routing all one-wire circuit conductors as close as possible to the conductive structure carrying its return current.

8.2.2 Distribution Control Unit

The distribution control unit controls and routes primary power to all the spacecraft subsystems and provides redundant engine and other control signals. The unit receives primary power from the power subsystem at a high current level through a high-reliability circular miniature connector containing large contacts. This input power is delivered over a double pair of large wires to achieve redundancy. Either pair can carry the full expected current in case of failure in the other pair. This unit does not need redesign for the upgraded spacecraft configuration. A small amount of excess capability is provided when the unit is used in the recommended spacecraft.

The distribution control unit contains connections to and from the in-flight jumper and also power switching relays for control of the several subsystem primary power buses. A simplified functional diagram illustrating this power control concept is shown in Figure 8-8. The power switching relays are connected in a redundant manner and can be switched off only by ground control through the test connector. They can be switched on by ground control through the test connector and also, as a backup, one decoder command will be assigned for the purpose of switching all power control relays on together in case one or more should somehow bounce to the off position. The benefits of this system are:

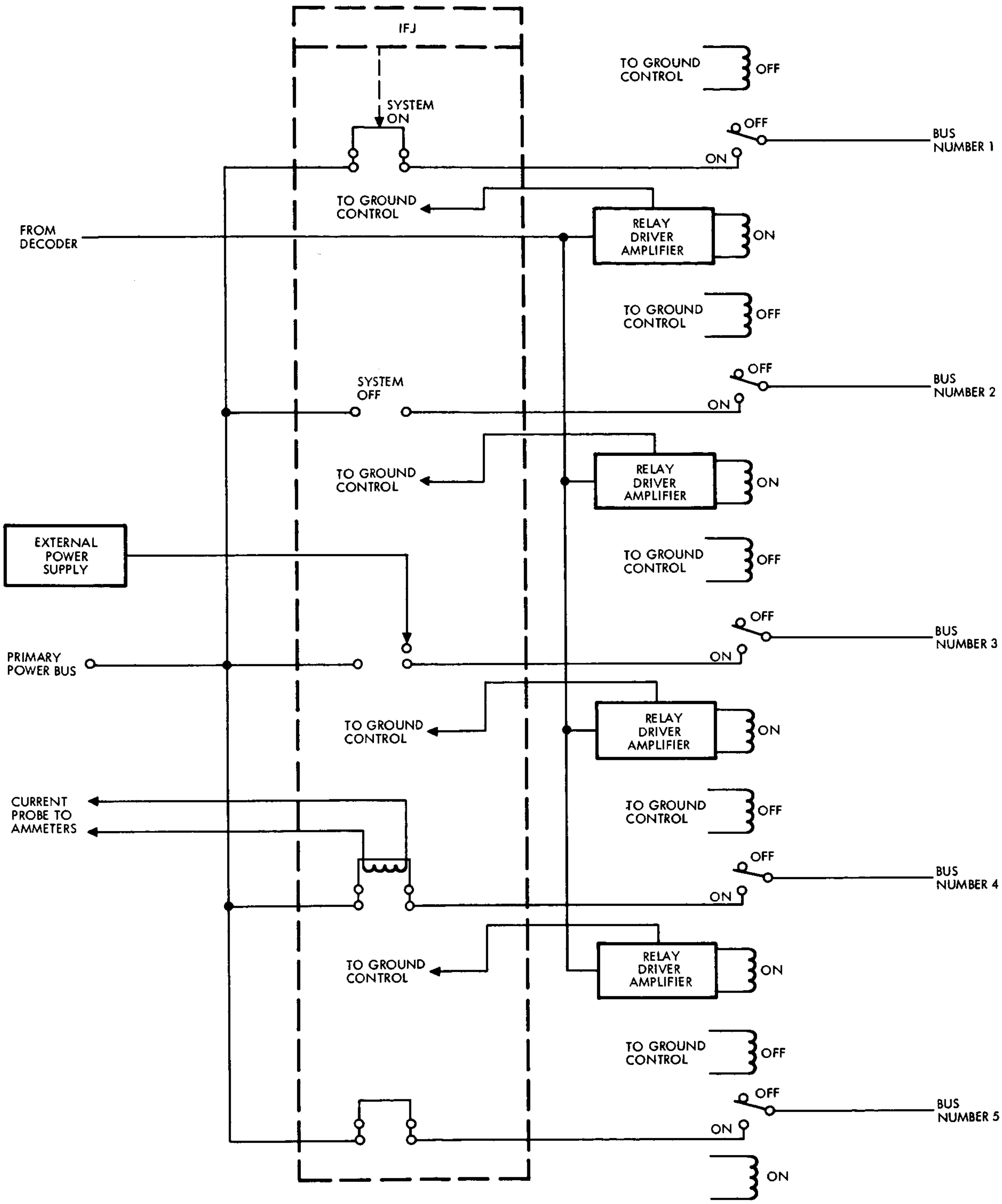


Figure 8-8
THE DISTRIBUTION CONTROL UNIT switches power to the different subsystems on command.

- During integration and test activities the power required by individual buses can be measured at the in-flight jumper.
- Individual bus power can be supplied by external sources during integration and test.
- During launch operations (spacecraft on stand) the individual buses can be switched off and on by direct wire control through a test connector.
- The decontaminated spacecraft need not be disturbed (by removing and inserting the in-flight jumper) to switch power off and on.

For certain engine control functions, time delay requirements are such that contact transfer times cannot be tolerated. For these functions, solid state circuits replace relays. Examples are:

- Open/close engine valves (high thrust)
- Open/close engine valves (low thrust)

Other relays are included for control of power to various units such as the capsule and the UHF data link.

8.2.3 Pyrotechnic Control Unit

This unit provides arming and firing control functions for all spacecraft ordnance devices. It has a "master safe-arm" series function (controlled by PVA/spacecraft separation) followed by an "arm" function (controlled by commands) followed by a "firing" function (also controlled by commands). All relays in this unit produce a short circuit across their output terminals when in the "safe" condition.* Figure 8-9 illustrates this concept. The "master safe-arm" cannot be put in the "safe" position after launch. It can be "safed" only on the ground prior to launch through a system test connector. This is a launch requirement and if this relay is armed, the countdown must stop.

*Considerations affecting the selection of relay firing circuits or solid state circuits and EED hot wire versus EBW are given in Appendix U.

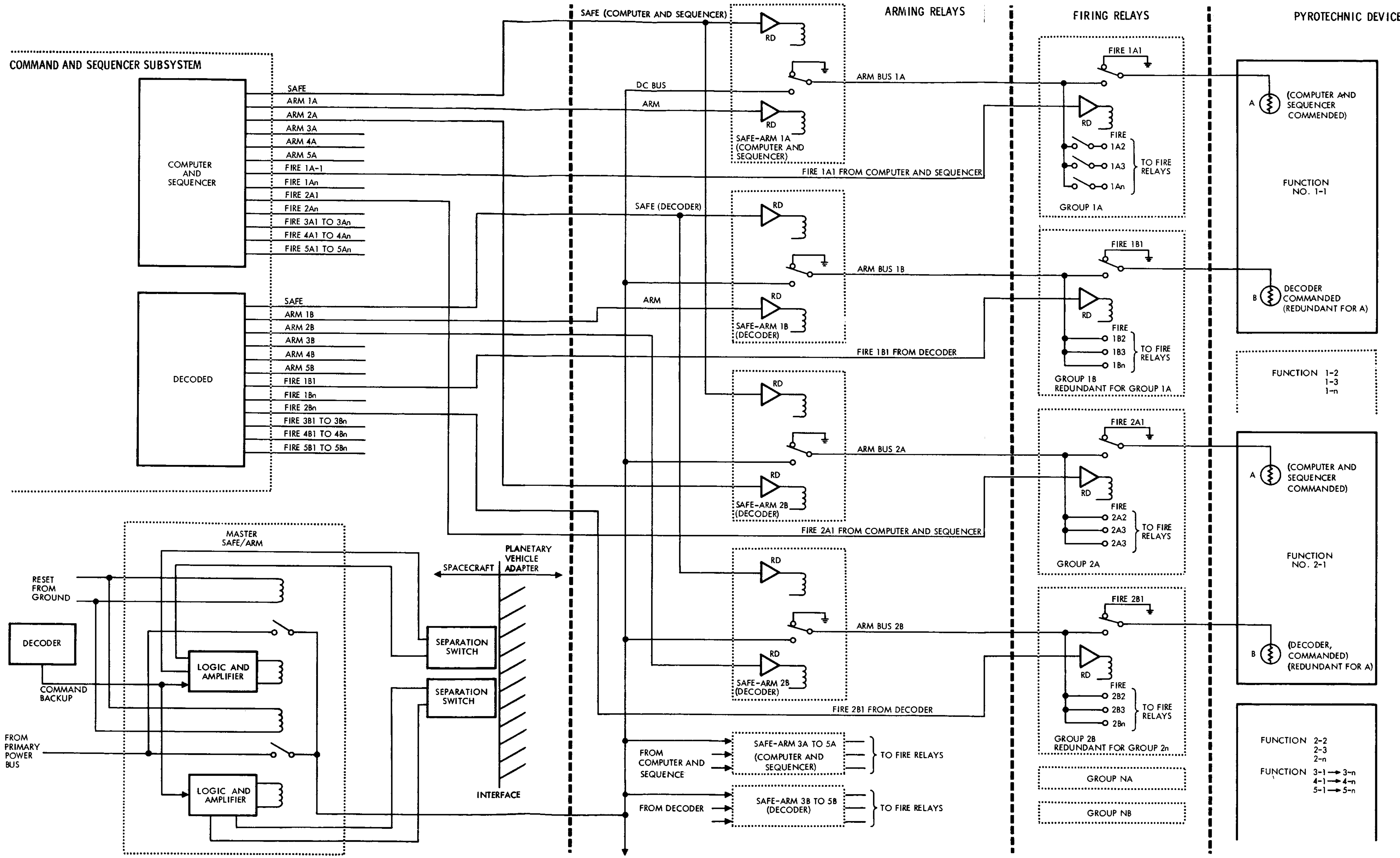


Figure 8-9

THE PYROTECHNIC CONTROL SYSTEM makes use of proven relay techniques in the master safe/arm, arming, and firing circuits. Commands originate from the spacecraft computer and sequencer or from the ground backup via the command subsystem.

A-13

FOLDOUT FRAME

FOLDOUT FRAME

A block diagram illustrating the complete ordnance firing circuitry concept in the pyrotechnic control unit is shown in Figure 8-9. The unit is suitable for the recommended 1973 spacecraft and the upgraded spacecraft. A function list for this unit is shown in Table 8-1.

8.2.3.1 Master Safe-Arm

Arming and firing cannot occur until after separation has occurred between the spacecraft and adapter. Switches sense the presence of the adapter and initiate a signal to the pyrotechnic control unit when the adapter is gone. This signal causes the master safe-arm relay in the unit to operate, thus making electrical power available to the arming relays (Figure 8-10). A backup command from the decoder can be used to operate the master arm function in case of failure in the separation switch circuits.

8.2.3.2 Safe-Arm Sections

Ten separate arming circuits are supplied for five double groups of firing relays in the firing control sections. Five arming circuits are triggered by commands from the computer sequencer, and the redundant (backup) groups are triggered by commands from the decoder. The 10 arming circuit outputs are delivered to the firing control section of the unit. A block diagram for this function is shown in Figure 8-9.

8.2.3.3 Firing Control Section

The firing control section of the unit contains 44 output circuits arranged to provide true redundant control for 22 functions. These include several groups for engine control, antenna control, and planetary scan platform deployment. The power for producing firing signals is derived from the several output circuits of the safe-arm section. A block diagram for this function is presented in Figure 8-9.

8.2.3.4 Pyrotechnic Circuitry EMC Considerations

Pyrotechnic circuits are routed separately from all other spacecraft wiring. The firing circuit wires, including electro-explosive device leads, are twisted to maintain electrical balance and minimize inductive pickup. Continuous circumferential shielding, grounded to structure at both ends,

Table 8-1. Pyrotechnic Control Unit Function List

Item	Function	Command Origin C and S	Command Origin Decoder	Command Origin Other	No. of Explosive Devices(1)	Arm Relay Designation Number
1	Safe group A	X				
2	Safe group B		X			
3	Pre Arm		X	Separation Switch		
4	Arm Relay 1A	X				
5	Arm Relay 1B		X			
6	Arm Relay 2A	X				
7	Arm Relay 2B		X			
8	Arm Relay 3A	X				
9	Arm Relay 3B		X			
10	Arm Relay 4A	X				
11	Arm Relay 4B		X			
12	Arm Relay 5A	X				
13	Arm Relay 5B		X			
14	Low gain antenna 1 and 2 deploy A	X			2	1A
15	Low gain antenna 1 and 2 deploy B		X		2	1B
16	High gain antenna release A	X			1	1A
17	High gain antenna release B		X		1	1B
18	Medium gain antenna release A	X			1	1A
19	Medium gain antenna release B		X		1	1B
20	Open He pressure valve 1st M.C.A.	X			1	2A
21	Open He pressure valve 1st M.C.B		X		1	2B
22	Pressurize tanks 1st M.C.A	X			2	2A
23	Pressurize tanks 1st M.C. B		X		2	2B
24	Close off pressure valves 1st M.C. A	X			3	2A
25	Close off pressure valves 1st M.C. B		X		3	2B
26	Open He pressure valve orbit A	X			1	3A
27	Open He pressure valve orbit B		X		1	3B
28	Pressure tanks orbit A	X			2	3A
29	Pressure tanks orbit B		X		2	3B
30	Close off pressurization valves (lockup) A	X			3	3A
31	Close off pressurization valves (lockup) B		X		3	3B
32	Depressurization He and fuel tanks A	X			2	4A
33	Depressurization He and fuel tanks B		X		2	4B
34	Depressurize oxidizer tank A	X			1	4A
35	Depressurize oxidizer tank B		X		1	4B
36	Planetary scan platform release A	X			1	5A
37	Planetary scan platform release B		X		1	5B
38	Open C-1 engine fuel and oxidizer valve A	X			2	2A
39	Open C-1 engine fuel and oxidizer valve B		X		2	2B
40	Spare encaging stop A	X			1	
41	Spare encaging stop B		X		1	
		20	21		46	

(1) Functions that require more than 1 EED, are fired simultaneously.

and containing no electrical discontinuities, is utilized from the explosive device to the point at which the firing circuit leads are shorted together. The cable shielding, electro-explosive device case shielding, and the isolation between ground monitor and control circuits and the firing circuits is at least 40 decibels from 150 kHz to 40,000 MHz as a design goal. The explosive devices are installed with a metal-to-metal contact resulting in a bonding impedance of less than 2.5 milliohms DC resistance and 80 milliohms RF impedance over the frequency range of 200 kHz to 20 MHz as a design goal. (See Figure 8-10.) The primary power input to the ordnance control box is filtered to prevent the conduction of interference into or out of the control box. This filtering eliminates the need to shield the primary power cable back to the power source, as well as protecting the pyrotechnic devices from interference present on the primary DC bus.

Cartridge design goals consider all Voyager preflight and flight environment, range safety requirements as applicable to AFETRM 127-1 category "A" devices, and the use of this one type in all Voyager electro-explosive device applications.

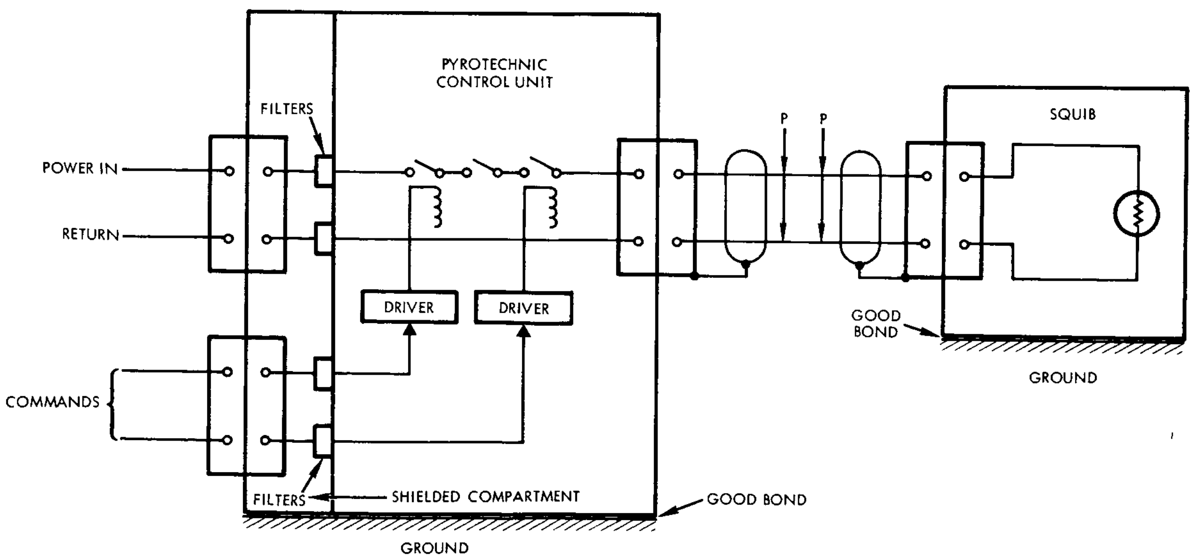


Figure 8-10

THE ORDNANCE SHIELDING CONCEPT IN THE PYROTECHNIC SYSTEM includes good bonding to the ground plane for all compartments, circumferential shielding, and input filters to the pyrotechnic control unit.

8.2.4 Reliability Assessment

The reliability block diagram for the EDAPCS is shown in Figure 8-11. The figure shows that the subsystem incorporates complete redundancy for all pyrotechnic control signals and primary power distribution.

The eight electrical distribution circuits associated with the equipment heaters and propulsion heaters are completely redundant, since failure of any one of the eight heaters is a system failure. Also, the safe-arm and pyro control circuits are redundant as failure of pyro devices, either premature firing or failure to fire, must be considered a system failure. Figure 8-11 shows that the pyro safe-arm and control equipment is divided into four major groups. Group 1 consists of the pyro equipment required to release the science and antenna booms. These booms are released no later than 10 days after launch. Group 2 consists of the pyro equipment required to release engine valves and solenoids for the first midcourse correction. This equipment is activated approximately 30 days after launch. At the time of Mars orbit insertion the third group of pyro devices is activated, prior to deboost engine firings. This maneuver occurs approximately 220 days after launch. Sometime after orbit insertion the fuel tanks are depressurized, requiring firing of additional pyro devices. The final group of pyro devices are activated to deploy the science experiments. This occurs when the spacecraft is in orbit around Mars.

8.3 COMPONENT DESCRIPTION

8.3.1 Harnesses

The harnesses use solder type and crimp type D series subminiature and circular miniature connectors throughout. A great deal of successful fabrication and flight experience has been gained with these connectors at TRW.

The "D" subminiature connector has high reliability, low cost, and easy assembly and maintenance compared with other high reliability subminiature connectors. These connectors perform very well for all normal command, control, telemetry, power and test signals. For ordnance circuits, connectors capable of circumferential shield termination are used.

The table below lists the EDAPCS equipment, the failure rate, the probability of success for the mission per unit, and the probability of success for the redundant groups. Since the equipment is required to operate up to a specific time, five time intervals are required. Group 1 is required to operate up to 240 hours; Group 2 up to 720 hours; Group 3 up to 5360 hours; and Group 4 up to 5500 hours. The electrical distribution equipment is required to operate for the full 6800 hour mission.

TABLE I

EDAPCS EQUIPMENT

Name	Failure Rate (λ) (bits per 10^9)	Probability of Success $e^{-\lambda_i T_i}$	Probability of Success Redundant Configuration
1. Electrical Distribution	1650	0.9888	0.9999
2. Pyro Group 1 - One Safe-Arm Unit and seven pyrotechnic control units	5850	0.9986	0.9999
3. Pyro Group 2 - One Safe-Arm Unit and three pyrotechnic control units	3450	0.9975	0.9999
4. Pyro Group 3 - Two Safe-Arm Units and nine pyrotechnic control units	8700	0.9544	0.9979
5. Pyro Group 4 - One Safe-Arm Unit and three pyrotechnic control units	3450	0.9812	0.9997
COMPLETE SUBSYSTEM			0.9966

The assumptions employed in performing the analysis are:

1. Mission time for electrical distribution equipment is 6800 hours
2. Mission time for pyrotechnic equipment is:
   a. Group 1 – 240 hours
   b. Group 2 – 720 hours
   c. Group 3 – 5360 hours
   d. Group 4 – 5500 hours
3. Times to failure are exponentially distributed. In the case of relays that are required to operate only one time or a few times, it may be required to replace the assumed constant failure rate by a cyclical failure rate. This will be the topic of future study.
4. This assessment assumes that the harnesses and J-boxes have reliability numbers of 0.99999

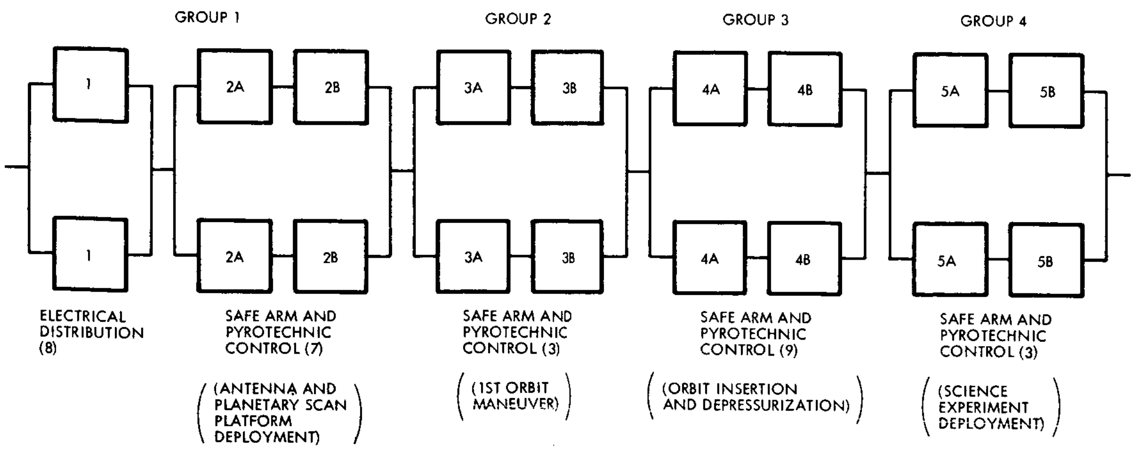


Figure 8-11

RELIABILITY ASSESSMENT OF THE ELECTRICAL DISTRIBUTION AND PYROTECHNIC CONTROL SYSTEM shows that the overall subsystem reliability is 0.9966.

For power circuits requiring very large currents, circular miniature connectors will be used. TRW has parts specifications for high reliability versions of circular miniature connectors which are now being used on other TRW space programs.

The electrical wire used in distribution hardware (harnesses and junction boxes) has lightweight insulation compatible with the space environment and flat braid shield. This gives a minimum weight final product with very good environment resistance and minimal fabrication effort. Small gauge wire will be used where the voltage drop is allowable. TRW has extensive experience with use of such wire on other spacecraft programs.

Devices that make the electrical connection to the planetary scan platform are supplied as part of the cabling. These devices accommodate about 180 degrees rotation of the platform in each of two axes. They are made up of standard circular wire conductors formed into flat cables stacked in layers and formed in a spiral. This concept is illustrated in Figure 8-16. Other cable flex devices using more complex concepts have been developed at TRW and are now in use on other spacecraft programs. These flexing cables will require thermal protection in the form of a bellows type jacket and a small heater.

In general, protective jacketing is not used on harness assemblies. Instead, the bundles are held together by string ties. This gives minimum weight, and has been well proven in other space programs at TRW. (Figure 8-17.)

The umbilical connectors between the spacecraft and capsule and between the spacecraft and launch vehicle are explosively separated. Circular miniature pygmy types successfully used on other TRW spacecraft programs are recommended. An alternate type requiring no separation force or action is also available.

All harness connectors are potted with semi-flexible compounds to provide wire-to-wire insulation resistance, protect against entry of contaminants which can cause shorting or arcing, retain contact float characteristics and provide increased and stress relief. Extensive experience with this technique has proved the reliability of the approach.

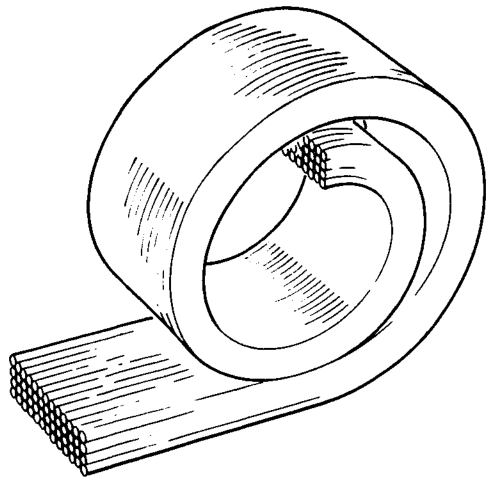


Figure 8-12
THE FLEXIBLE CABLE CONNECTION to the planetary scan platform is relatively simple compared with other flexible connection schemes developed at TRW to meet more demanding requirements in total rotational travel and cable complement.

8.3.2 Junction Boxes

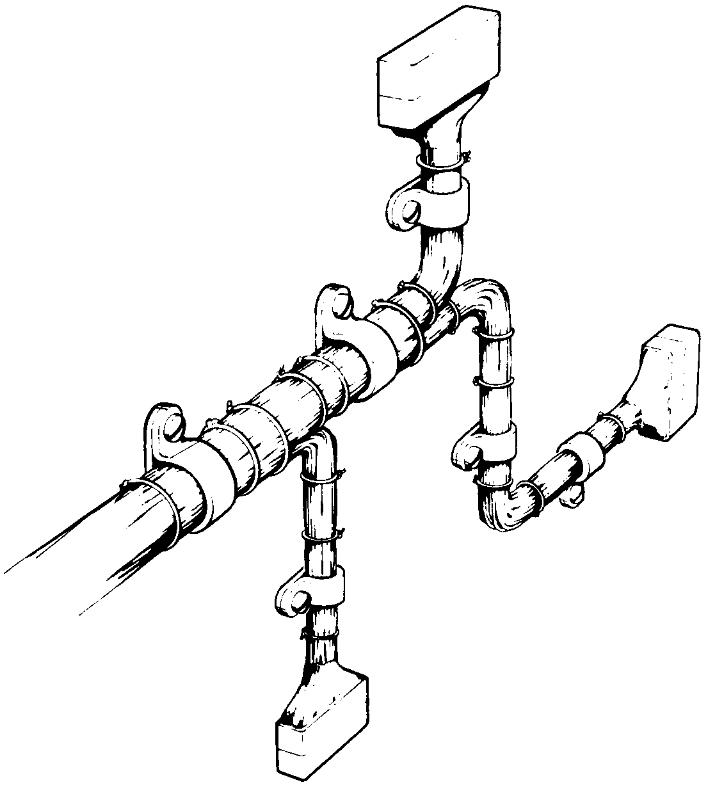
The junction boxes are mostly rectangular containers of sheet aluminum, chemically treated to produce a gold iridite finish. This conductive finish satisfies the system EMC requirements. It contacts the conductive finish on the unit connectors, grounding the connector shells so that they act as a continuous shield. Harnesses and terminal boards inside the junction boxes meet the same requirements as external interconnecting harnesses. Terminal boards are conventional rectangles made of epoxy fiberglas mounted with standard terminals that are flight approved. It is often possible to place 10 or more terminals on each square inch of surface area and readily accommodate reasonably sized conductors. J-boxes dissipate very little power and so do not have internal thermal design problems. A typical J-box sketch is illustrated in the preliminary specification of Figure 8-16.

Experiment power control relays will be housed in the science J-boxes. Flight qualified relays have been used extensively by TRW for this application on previous spacecraft programs.

The science J-box will also contain devices to protect the primary power system against excessive current drain by an experiment. It is planned that a resettable circuit breaker type of device will be developed

PRELIMINARY SPECIFICATION

Harness Segment



Purpose

To transmit electrical signals and power throughout the spacecraft for the various subsystems.

Performance Characteristics

Distribute electrical signals and power throughout the spacecraft in a manner such that EMC requirements are satisfied and voltage drop along any power line is minimized.

Physical Characteristics

Weight

Equipment Module Harness	206 Pounds
Propulsion Module Harness	5 Pounds
Planetary Vehicle Adapter Harness and Umbilical	8 Pounds
Equipment Module Umbilical	1 Pound
Total	220 Pounds

Size

Minimum

Operating temperature range

-30°F to +150°F for non-flexing elements. Flexing elements require thermal control limiting temperature to 70° ± 30°F.

Jacketing

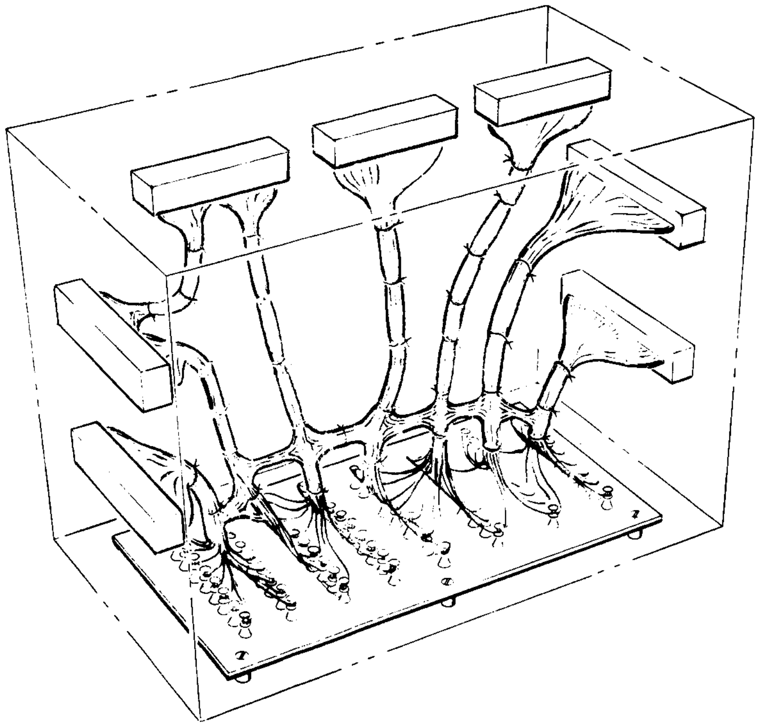
None

Reliability

0.9999

Figure 8-13

Typical J-Box



Purpose

To provide a protected mounting location for:

a) terminating the various subsystem harnesses, thereby maintaining harness modularity

b) electrical branch points (terminals) and electrical components (resistors, capacitors, diodes, fuses, etc.) as required for subsystem electrical signals.

c) experiment power control relays.

Physical Characteristics

Weight

Subsystem Equipment Panel J-Boxes (4 each)	28	Pounds
Science Equipment Panel J-Box (1 each)	3.0	Pounds
Propulsion Module J-Box (1 each)	5	Pounds
Planetary Scan Platform J-Box	13.0	Pounds
Total	49	Pounds

Size

Subsystem Equipment Panel J-Boxes	(4 each)	9 x 6 x 6
Science Equipment Panel J-Box	(1 each)	4 x 5 x 6
Planetary Scan Platform J-Box	(1 each)	9 x 4 x 12
Propulsion Module J-Box	(1 each)	6 x 12 x 3

Operating temperature range

+5°F to +150°F.

Finish

Conductive

Reliability

0.9999

Figure 8-14

and used in this application. An alternative is to use fuses or isolation resistors proven on previous TRW spacecraft programs.

8.3.3 Distribution Control Unit

The distribution control unit has an input connector of a circular miniature type accommodating large contacts for high currents. The output connectors are appropriate subminiature "D" series units. All output circuits are double wired to provide redundancy. The unit external housing has a fully conductive gold iridite finish. A preliminary specification sheet for this unit is presented in Figure 8-15.

8.3.4 Pyrotechnic Control Unit

Power for the relay drivers is developed and regulated within the pyrotechnic control unit by redundant circuitry operating from raw battery bus voltage. The unit external housing has a fully conductive gold iridite finish. All connectors will be of the subminiature "D" series. Incoming commands will all pass through filters to maintain the unit shield integrity.

The relay driver amplifiers throughout this unit cause relay coil actuation current to flow only for about 100 milliseconds. This allows the regulated power supplies to be relatively small in size and minimize heat resulting from power dissipation.

A preliminary specification sheet for this unit is shown in Figure 8-16.

The 11 relays in the safe-arm section of the pyrotechnic control unit are crystal can size, as used on other TRW space programs. Such units can pass but not switch currents up to 20 amperes. These relays are of the latching type so they can properly respond to the short duration actuation pulses mentioned above.

The 44 relays in the firing control section of the pyrotechnic control unit are also crystal can size, as used on other TRW space programs. Such units can switch currents up to 10 amperes. This is compatible with the 5 ampere all-fire current requirement of presently planned squibs.

8.3.5 Initiator Cartridge

The preliminary specification for the initiator cartridge is given in Figure 8-17.

PRELIMINARY SPECIFICATION

Distribution Control Unit

Performance Characteristics

Input: 37-50 volts DC

Console signals from ground test connector. Commands from computer and sequencer and command subsystem.

Outputs (Switched):

Open Start Tank Valves	(400w)
Open/Close Engine Valves (Low Thrust)	(400w)
Open/Close Pre-Valves	(100w)
Open/Close Engine Valves (High Thrust)	(65w)
High Gain Antenna Hinge Motor Drive Thermal Control ON/OFF	(5w)
High Gain Antenna Shaft Motor Thermal Control ON/OFF	(5w)
Planetary Scan Platform Gimbal Motor Thermal Control ON/OFF	(5w)
TVC Power ON/OFF	(1300w)

(7 redundant sets of output circuits) for the seven switched subsystems.

Purpose

Provides distribution and switching of electrical power to spacecraft subsystems.

Physical Characteristics

Size: 7 x 7 x 10 inches

Weight: 12 pounds

Reliability

Reliability: 0.9999

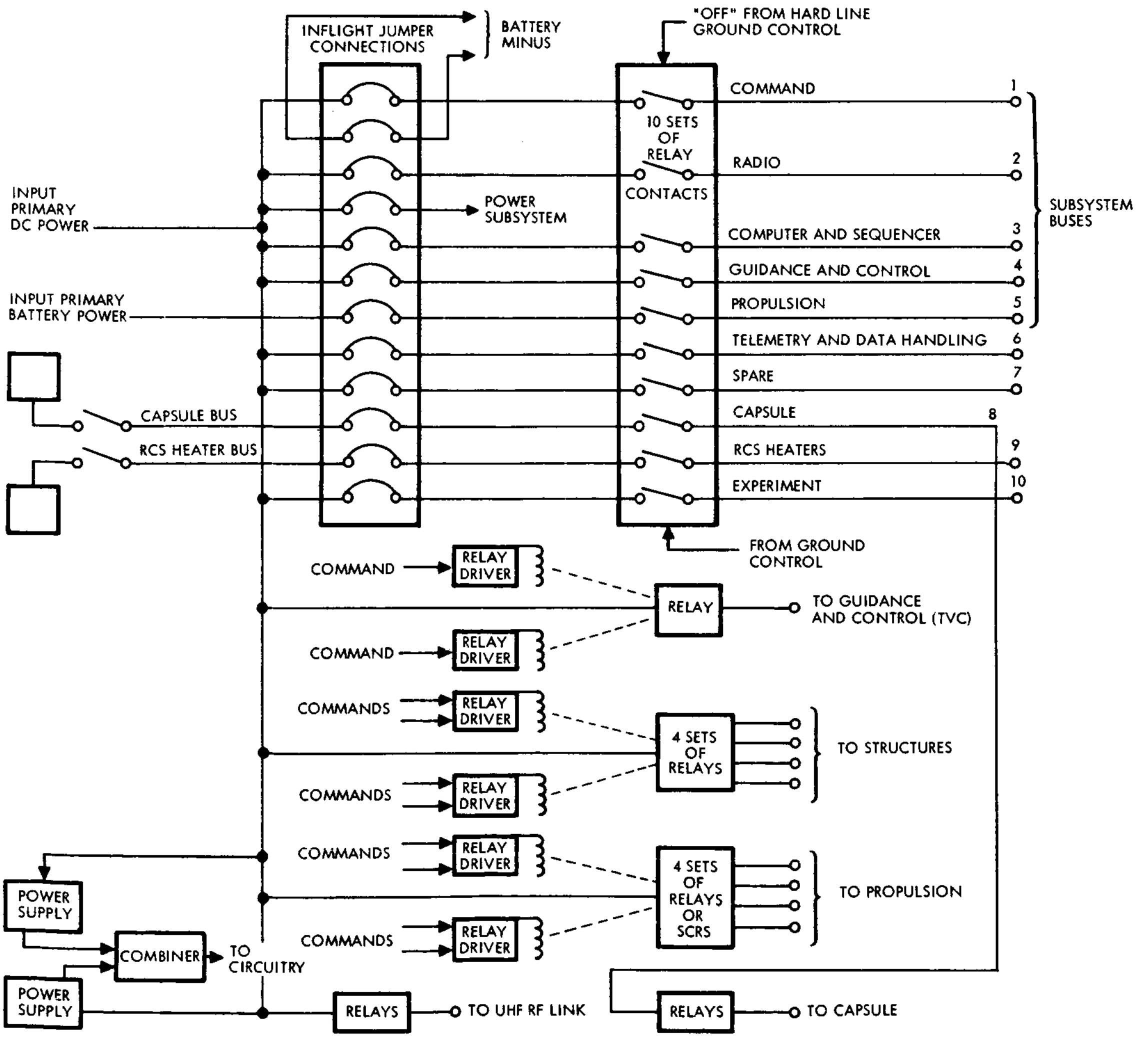


Figure 8-15

PRELIMINARY SPECIFICATION

Pyrotechnic Control Unit

Purpose

To provide circuitry to actuate explosive devices as required by mission objectives.

Performance Characteristics

SAFE-ARM:

Includes a pre safe-arm circuit and 5 separate safe-arm circuits to provide arming in conjunction with mission timing requirements.

Firing Current:

Battery direct.

Circuit Characteristics:

Firing Control: Relay, solid-state driver.
Safe-Arm: Latching Relay, solid-state driver.
Power Supply: Solid-state (redundant).

Input Voltage:

37-50 VDC.

Power Requirements:

0 watts (safe)
2 watts (armed)
8 watts max 50 msec (fire)

Command Inputs:

20 from computer and sequencer
21 from Decoder
pre-Arm Command from separation switch

Number of Controlled Explosive Device Outputs:

44 (dual redundant)

Number of TLM Outputs:

1 analog
11 discrete

Size:

7 x 7 x 15

Weight:

25 pounds

Reliability

0.997

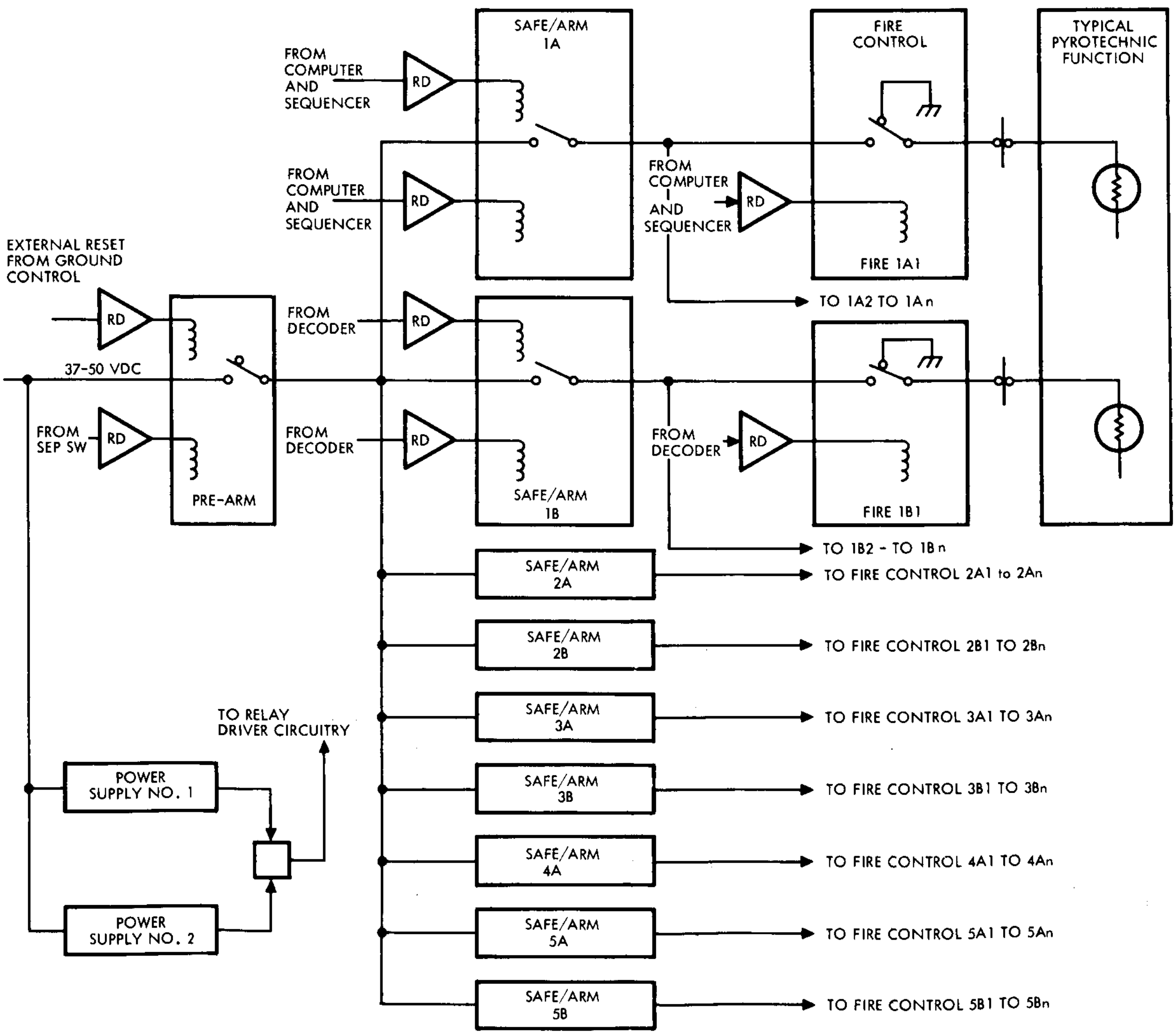
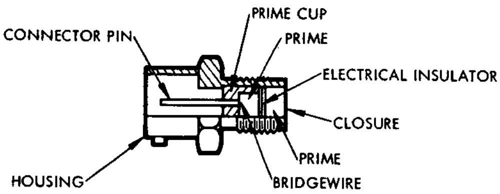


Figure 8-16

PRELIMINARY SPECIFICATION

Initiator Cartridge



INITIATOR CARTRIDGE

PRELIMINARY SPECIFICATION SHEET
INITIATOR CARTRIDGE

Purpose

To actuate pressure controlled devices

Type

Single bridgewire

Performance Characteristics

All fire current:	4.5 amp maximum for 0.020 seconds
No fire current:	1.0 amp or 1.0 watt for 5 minutes
Satisfy AFETRM 127-1 (for cat. "A" devices)	
Temperature Range:	-65°F to +250°F
Bridgewire Resistance:	1.0 ± 0.5 ohms
Insulation Resistance:	2 megohms minimum at 500 ± 25 VDC applied between shorted pins and case.
Static Discharge Sensitivity (No Fire):	25000 volts from a 500 ± 50 PICOFARAD capacitor applied between shorted pins and case.

Physical Characteristics

Material Housing - Stainless Steel
Contact pins - Stainless Steel

Reliability

0.996

Figure 8-17